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TURBOJET PROPULSION-SYSTEM RESEARCH AND THE RESULTING EFFECTS ON AIRPLANE PERFORMANCE

By Addison M. Rothrock

NACA Headquarters
Washington, D. C.

CLASSIFICATION CHANGED

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1 AT 9011

By authority of NACA Research Abstracts Date June 27, 1956
And Reclassification Notice No. 112 - effective date June 12, 1956. By: Jon B. Elliott

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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By Addison M. Rothrock

March 1955

Page 70, figure 17: Change the "-" to "+" to read:

$$W_{eng} = 1000 + 0.0890\ d_c^3 \text{ (Including AB.)}$$

[redacted] Fig. Corrected 4-29-55. JKE

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

TURBOJET PROPULSION-SYSTEM RESEARCH AND THE RESULTING EFFECTS ON AIRPLANE PERFORMANCE

3460

By Addison M. Rothrock

INTRODUCTION

For a period of ten to fifteen years intensive research and development has been conducted on turbojet propulsion systems for aircraft. During this period much has been learned about the systems both from the standpoint of current usage and of future development possibilities. It is the purpose of this report to discuss the current status of the turbojet engine as produced in the United States and to discuss the future possibilities for improvement in the engine and in the fuel. The engine and fuel improvements will be evaluated both from the standpoint of probability of success in obtaining these improvements and from the standpoint of the effects of these improvements on the airplane performance.

In considering the resulting advances in airplane performance, a further comparison will be made of the extent to which improvement in factors of the airplane not in the purview of the propulsion system designer will result in equal or better improvements in aircraft performance. It will be preferable before examining the advances that can be made in the propulsion system to determine these relative effects.

There are seven major propulsive-system factors that affect airplane performance. These are: (1) heat of combustion of the fuel, (2) density of the fuel, (3) efficiency of the engine, (4) specific weight of the engine, (5) specific area (square feet of frontal area per pound of thrust) of the engine, (6) stress limitation of the engine in terms of maximum permissible pressure loading of engine parts, and (7) maximum ambient temperature at which the engine can operate satisfactorily. For the range of airplane performance covered in this report and for the fuels considered, fuel-density (which determines the volume occupied by a given fuel weight) effects are of secondary importance and will not be considered. The effect of engine specific area on airplane performance is dependent on the installation of the engine in the airplane and of itself may or may not affect the airplane performance. Since this effect is dependent on the particular airplane design, it will not be considered in relating the engine performance to the airplane performance.

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Three major airplane factors that do not come under the purview of the propulsion-system designer will be considered. These are: (1) lift-drag ratio of the airplane, (2) percentage of gross airplane weight that is military load (defined herein as weight of the pilot, the armor, the armament, and the guidance system), and (3) percentage of gross airplane weight that is airframe (defined herein as gross weight of the airplane less military load, installed engine or power plant weight, and fuel weight).

The 10 airplane and propulsion-system factors considered together with the symbols used are:

L/D	lift-drag ratio of the airplane (considered herein as the L/D ratio of the trimmed airplane in level flight at the altitude and speed under consideration)
W_f/W_g	ratio of fuel load to gross weight
W_m/W_g	ratio of military load to gross weight
W_{af}/W_g	ratio of airframe weight to gross weight
W_e/W_g	ratio of installed power plant weight to gross weight
h	heat of combustion of the fuel
η_e	over-all efficiency of the engine, that is, the ratio of work done on the airplane (thrust time distance flown) to thermal energy of the fuel consumed in flying the distance
W_{eng}/F	specific engine weight, which is the ratio of engine weight to thrust produced
ΔP_{max}	maximum permissible engine pressure loading (such as hoop stress)
T_{max}	maximum permissible stagnation (total) temperature

For a specific airplane, certain of these variables remain constant regardless of airplane mission: heat of combustion of the fuel h, weight of the airframe W_{af}, weight of the installed power plant or engine W_e or W_{eng} maximum permissible engine pressure loading ΔP_{max}, and maximum permissible stagnation temperature T_{max}. For a particular mission, military load can be considered constant over considerable portions of the flight. The remaining factors vary during a flight. Fuel weight W_f and gross weight W_g decrease continuously as fuel is consumed. Engine efficiency η_e, thrust produced F, and airplane drag D vary during the flight, but in no preordered manner. Each of these three parameters is at any instant a function of the instantaneous values of airplane speed, altitude, [redacted]hange of altitude.

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Since airplane drag, that is thrust required, varies with the flight condition, L/D also varies with the flight condition. If the airplane is flying in level flight at a constant speed, the lift is equal to the airplane weight, or $L/D = W_g/F_{required}$. If the airplane is climbing or accelerating, the thrust required is increased by the amount required to produce the rate of climb or the acceleration.

Specific engine weight at any flight condition is considered to be engine weight (a constant) divided by the thrust produced at that flight condition; consequently specific engine weight varies inversely as the thrust. When the thrust produced is the full-throttle thrust of the engine, specific engine weight is a minimum for the airplane speed and altitude under consideration. In this analysis, this minimum specific engine weight is generally the value of interest. Obviously, once a specific airplane is considered, thrust available and the thrust required (equal to airplane drag plus the thrust required for acceleration or climb) are the factors of interest rather than specific engine weight or airplane lift-drag ratio. Since the purpose of this discussion is to interrelate the basic engine and airframe characteristics to generalized airplane performance, and not to consider the performance of specific airplanes, the relationships of the variables listed must be examined.

The effects on over-all airplane performance of the factors listed will be analyzed by examining their effects on airplane gross weight, range, altitude, and speed. In addition, the relation of engine size to airplane gross weight will be discussed. Although the examples given are for a turbojet-powered aircraft, the method of analysis is equally applicable to airplanes powered with any type of air-breathing engine.

After this analysis, turbojet engines currently being manufactured in the United States are examined with respect to values of η_e, W_{eng}/F and ΔP_{max}. Fuels are examined in relation to h and also, because the fuel determines combustion temperature obtainable (and thereby affects thrust) in relation to W_{eng}/F. The relation of stagnation temperature to engine development, with particular reference to lubrication, is discussed.

Except as affected by reserves of stored fuel, the total number of miles that can be flown per day with turbojet engines is a function of the production rate of turbojet fuel. The rate at which engines can be produced is dependent on the availability of the materials from which the engines are made. These two factors, availability of fuel and engine production limits as a function of material availability, are discussed. Engine reliability is not discussed in any detail in this analysis.

Preparation of this report required much specific engine data from the aircraft engine manufacturers. These data were supplied by the aircraft engine industry at the request of the Department of Defense. This cooperation on the part of industry and of the Department is appreciated.

Determination of the effects of the various independent variables required much analytical work beyond that included in this report. These analyses were made by staff members of the NACA Lewis laboratory and of the NACA Headquarters Office. Particular credit is due Richard S. Cesaro of the NACA Headquarters staff and E. Clinton Wilcox of the Lewis laboratory staff for their assistance. 3480

RELATIVE EFFECTS OF AIRPLANE AND ENGINE FACTORS

ON AIRPLANE PERFORMANCE

Effects on Range

The heat of combustion of a fuel is generally expressed as so many thermal units per pound (approximately 18,500 Btu per pound for hydrocarbon turbojet fuels). For the current analysis, this value is better expressed as the number of miles for which one pound of thrust is produced by burning one pound of fuel. Thus, if all the chemical energy in one pound of a typical JP-4 turbojet fuel were converted into thrust, it would produce one pound of thrust for approximately 2400 nautical miles.

Because an engine does not have an efficiency of 100 percent, the distance over which this pound of thrust is available is considerably less than the ideal value. For the operating conditions considered in this report, turbojet engine over-all efficiency ranges between 10 and 40 percent. Therefore, as used in the engines considered, one pound of JP-4 fuel produces one pound of thrust for a distance between 240 and 1000 nautical miles. If a fuel having a higher heat of combustion is used, these thrust-mile values increase proportionally. For current jet fuel, the relationship between engine efficiency η_e and specific fuel consumption sfc is η_e = (airplane speed in knots)/(2400×sfc) where sfc is in terms of pounds of fuel per hour per pound of thrust, or $\eta_e = 0.136/\mathrm{sfc}$ where sfc is in terms of pounds of fuel per hour per horsepower.

As stated, the value of $k\eta_e$ determines the airplane flight distance over which one pound of fuel will produce one pount of thrust. The number of pounds of airplane weight that this one pound of thrust will support in the air (W_g/F) is equal to the lift-drag ratio L/D of the airplane at the flight condition considered. The lift-drag ratio may be considered as an "efficiency" of the airframe in that its reciprocal D/L expresses the amount of work that must be done per pound of airplane in flying the airplane a given distance. For a given L/D, the work done in flying the airplane a given distance is independent of the speed at which the flight is made.

The airplane weight to be supported comprises the weights of airframe W_{af}, military load W_m, power plant W_e, and fuel W_f. The sum of these weights is the gross weight W_g of the airplane. If the value of $h\eta_e$, is multiplied by the airplane lift-drag ratio (L/D) and by that portion of the pounds of airplane that is fuel, (W_f/W_g), an approximate value for the range R is obtained:

$$R \doteq h\eta_e \frac{L}{D} \frac{W_{f_o}}{W_{g_o}} \tag{1}$$

In the equation L/D is considered constant. The subscript o indicates the values at the start of the portion of the flight under consideration.

Equation (1) is an approximation in that it does not consider the weight reduction that occurs during the flight as fuel is consumed and the consequent reduction in thrust requirement. This effect is accounted for in the classical Breguet range equation. For a fuel load of 10 percent of the gross weight, the range given by equation (1) is in error by 4 percent with respect to that given by the Breguet equation. As fuel load increases, this error increases rapidly.

The Breguet equation, with h, η_e, and L/D assumed to be constant, states that:

$$R = h\eta_e \frac{L}{D} \int_{W_{g_o}}^{W_{f_o}} - \frac{dW}{W} \tag{2}$$

$$R = h\eta_e \frac{L}{D} \log_e \frac{W_{g_o}}{W_{g_o} - W_{f_o}} \tag{3}$$

All the terms in equation (3) are in equation (1), but equation (3) provides the correction necessary to account for the effect on range of the continually decreasing fuel load.

To estimate the ranges that are feasible with current military aircraft, equation (3) is evaluated using appropriate values for the different variables. Two cruise speeds, $M_a = 0.9$ and $M_a = 2.0$, will be considered. It will be assumed that the engine afterburner is not required at the lower speed, but is at the higher speed. Over-all engine efficiency, as will be explained later, is about the same under these two conditions; a value of 0.22 is representative of current practice. For a bomber designed to cruise at $M_a = 0.9$ that does not have supersonic capabilities, an L/D value of 20 approximates current design values.

For a bomber designed to cruise at $M_a = 2.0$, an L/D value of 5 is assumed. For the fighter, the corresponding lift-drag ratios are assumed to be about 60 percent of these values. The value of h, as previously stated, is 2400 nautical mile-pounds of work per pound of fuel. With these data, the radius of action (1/2 R) is computed from equation (3) for different values of W_f/W_{g_o}, figure 1. For a long-range bomber, the fuel weight available for cruise is about 0.50 of the airplane gross weight. At $M_a = 0.9$ (L/D of 20), the radius of action of such an airplane is, therefore, without flight refueling, about 3500 nautical miles. 3460
At a flight speed $M_a = 2.0$ (L/D of 5), the range is 750 to 900 nautical miles. If the cruise portion of the flight is at $M_a = 0.9$, and there is a $M_a = 2.0$ supersonic dash over the target, the required design compromises will result in lower lift drag ratios in the subsonic speed range than that used in preparing figure 1.

With current interceptors, the fuel available for cruise is about 0.15 of the gross weight. The combination of this lower percentage of fuel and the lower lift-drag ratios of the fighter results in cruise radii of about one-seventh the values estimated for the long-range subsonic bomber.

The design of a military airplane is a compromise between the somewhat counter objectives of low gross weight (considering both the pounds of airplane and of fuel) and high aircraft performance. The way in which the compromise is made determines the manner in which gross weight is divided into airframe, military load, power plant, and fuel. For this reason, equation (3) will be modified to express the individual effects of airframe weight, military load, and specific engine weight on airplane range.

Equation (3) is rewritten by substituting for W_{f_o} the equivalent value $W_{g_o} - (W_a + W_m + W_e)$ and dividing numerator and denominator by W_{g_o}. Equation (3) then becomes:

$$R = h\eta_e \frac{L}{D} \log_e \frac{1}{\frac{W_{af}}{W_{g_o}} + \frac{W_m}{W_{g_o}} + \frac{W_e}{W_{g_o}}} \tag{4}$$

For the airplane in level flight at a constant speed:

$$\frac{W_e}{W_{g_o}} = \frac{W_e/F_o}{W_{g_o}/F_o} = \frac{W_e/F_o}{L/D}$$

Substituting in equation (4), the range is expressed as:

$$R = h\eta_e \frac{L}{D} \log_e \frac{1}{\frac{W_{af}}{W_{g_o}} + \frac{W_m}{W_{g_o}} + \frac{W_e/F_o}{L/D}} \tag{5}$$

Equation (5) shows the manner in which the major factors previously presented affect airplane range, h, η_e, and L/D being considered constant during the flight phase considered. For those cases in which variations in W_e/W_{g_o} are not being considered, equation (4) is preferred to equation (5).

In table I, current military aircraft are described according to the weight distributions made by aircraft designers. Representative values are listed for a bomber and for a fighter. There are deviations from these typical figures, but the deviations are not large enough to affect the conclusions drawn from this discussion. The installed power plant includes those parts which although not supplied by the engine manufacturer are attached directly to the engine and are required by the engine. This weight is about 25 percent greater than the weight of the engine as supplied by the engine manufacturer. Consequently, weight of the engine proper is 10 percent of the gross weight for the bomber and 20 percent for the fighter. In this discussion, weight of the installed power plant will be designated W_e and weight of the engine as supplied by the engine manufacturer W_{eng}. In this analysis it is assumed that any change in W_{eng} is accompanied by a proportional change in W_e.

TABLE I. - REPRESENTATIVE AIRPLANE WEIGHT DISTRIBUTION AT TAKE-OFF

	Bomber	Fighter
Airframe	30.0%	37.5%
Installed power plant	12.5*	25.0**
Military load	7.5	7.5
Fuel	50.0	30.0

*With or without afterburner.
**With afterburner.

The engine group is a higher percentage of gross weight for the fighter than for the bomber largely because L/D is lower for the fighter.

In figure 2 is shown the effect on airplane range, as determined from equation (4), of changing the proportions of military load and fuel comprising the weight allotted to military load plus fuel. The assumption is made that all the fuel is available for cruise. This assumption does not modify the general conclusion to be drawn. The weight of military load plus fuel selected for these curves is 57.5 percent of the gross weight for the bomber and 37.5 percent for the fighter. The curves apply to both a constant gross weight with varying military load or a constant military load with varying gross weight.

The military load (as defined herein) is usually between 1500 and 2000 pounds for current fighters and about ten times these values for long range bombers. For constant military load, (fig. 2) if the percentage of gross weight allotted to fuel is to be increased for greater range, the ratio W_m/W_{g_o} must be decreased by increasing W_{g_o}, the gross weight of the airplane. In figure 3, curves are plotted that show this effect. A constant military load of 2000 pounds for the fighter and 20,000 pounds for the bomber is assumed. The curves show that for either bomber or fighter a ratio of military load to gross weight of about 7.5 percent represents a reasonable balance between the objectives of long range and low gross weight. The curve shows the manner in which the gross weight of the airplane is in general determined by the military load. Certain conditions such as aircraft carrier size may impose a limitation on airplane gross weight that is independent of military load considerations. In other cases, the range required may be so low that the ratio of military load to gross weight can be considerably higher than the 7.5 percent figure. 3450

Equations (4) and (5) are now examined (figs. 4 and 5) to determine the effects on range of increasing fuel heat of combustion, engine efficiency, or airplane lift-drag ratio, or of decreasing the percentage of gross weight allotted to airframe or to power plant. In this comparison, no estimate is made of the practicability of achieving the improvements discussed. An increase in h or in η_e increases the distance over which a pound of fuel will produce the necessary thrust. An increase in L/D decreases the thrust requirement and thereby the rate at which fuel is consumed. A decrease in specific airframe or engine weight allows a greater percentage of gross weight to be allotted to fuel.

The equations show that an increase in either h or η_e increases range in direct proportion.

To estimate the effect of change in L/D, two cases are considered. In the first case, it is assumed that as L/D is increased, thereby reducing the thrust requirement, the percentage of gross weight allotted to the power plant is proportionally decreased (that is, for a constant value of W_e/F, the ratio of thrust available to the thrust required at any flight condition remains unchanged) and this weight increment can be added to fuel weight. As an example, if L/D for the fighter is increased 30 percent, the ratio of power plant weight W_e to gross weight W_g is decreased by $0.250 - \frac{0.250}{1.30} = 0.058$. This 5.8 percent of the gross weight may then be added to the fuel weight, which would thereby be increased to 35.8 percent of the gross weight. Such an increase in fuel weight would of itself increase range 24 percent, if airplane L/D remained constant (eq. (3)). The total range increase for the 30-percent increase in L/D is therefore $(1.30 \times 1.24 = 1.61)$ 61 percent.

In the second case, it is assumed that weight distribution of the airplane is unchanged, and equation (4) applies. An increase in L/D is therefore accompanied by a directly proportional increase in range. If the value of W_e/F is unchanged, an increase in the ratio of thrust available to thrust required also results, and the attainable altitude and speed are increased. The extent of these increases will be discussed later.

3460 It is assumed that weight saved by decreasing the ratio of airframe weight to gross weight W_{af}/W_{g_o} is allotted to fuel weight. That is, for the bomber, if airframe weight is decreased 20 percent (W_{af}/W_{g_o} decreased from 0.30 to 0.24), fuel weight is increased 12 percent (W_f/W_{g_o} increased from 0.50 to 0.56). The accompanying increase in range is determined from equation (3).

The effects of a decrease in specific engine weight are considered with the assumption that ratio of installed power plant weight to gross weight is decreased in proportion to any decrease in W_e/F; that is, the thrust available for any flight condition is not affected by the reduction in specific engine weight. The saving in power plant weight is used to increase fuel weight with an accompanying increase in range, as shown in the W_e/F curves of figures 4 and 5.

For either the bomber or the fighter, an increase in L/D accompanied by a proportional decrease in W_e/W_{g_o} increases range more than a change in any other parameter being considered. For the fighter, a decrease in W_{af}/W_{g_o} is almost as effective. For the bomber, a decrease in W_{af}/W_{g_o} is less effective because W_{af} is a lesser part of the total. Since the engine of a bomber is a small part of total airplane weight, specific engine weight has much the least effect on bomber range. With the fighter, the effect of a decrease in W_e/F_o is about equivalent to that of an increase in h or in η_e.

For either airplane, cumulative range extensions result from simultaneous improvement of more than one of the variables. Again, it is mentioned that although the results approximate the range benefits that can accrue from improvements in the five variables considered, the results in no way imply the extent to which these improvements can be obtained, nor do the results imply whether or not it is advisable to use the improvement in any one factor to improve range, rather than speed or altitude.

For the curves presented in figures 4 and 5, it has been assumed that the airplane flies at constant values of engine efficiency, specific weight, and airplane L/D. An actual flight, particularly that of the fighter, covers a wide range of speeds and altitudes with corresponding variations in η_e, W_e/F, [redacted]

One of several possible fighter or interceptor flight plans is shown in figure 6, together with a representative breakdown of the fuel load allotted for each phase of the flight. If the effects of any of the improvements shown in figure 5 are concentrated into one phase of the flight, the phase increase can be greater than the over-all values shown in figure 5. For instance, a 10 percent improvement in engine efficiency during the whole flight would mean a 10 percent saving in fuel. If all of this 10 percent saving in fuel is applied to the cruise phases, a 23-percent improvement in cruise range $\left(\frac{3}{13} \times 100\right)$ results. If it is applied entirely to the combat phase, a 75 percent improvement $\left(\frac{3}{4} \times 100\right)$ in combat 3460
time results. A decrease in the installed power plant weight from 25 to 20 percent of the fighter gross weight would, if this weight saving were used to increase fuel load, permit combat fuel to be increased from 4 percent of the gross weight to 9 percent, which would more than double combat time. In over-all planning of research programs, it is questionable if analysis of the effects of engine variables on airplane performance should be carried much beyond the present type of treatment.

Specific estimates can be made of the range expected with current airframes and propulsion systems and of the range extension that might result from a combination of improvements in airframe and engine performance. An example of such estimation is presented in figure 7. In this case, a long-range interceptor is considered with a weight distribution between those listed in table I for the bomber and for the fighter. The fuel allotted for each of the two cruise phases is 17 percent of the gross weight at the start of the phase. The fuel is assumed to be JP-4. In the figure, curves for constant engine efficiency, based on conventional engine operating conditions, are first plotted to show radius in miles (that is cruise out or cruise back) as a function of airplane lift-drag ratio. Since W_{f_o}/W_{g_o} is constant, these curves (see eqs. (3) or (5)) are straight lines. Three airplane cruise Mach numbers are assumed: 0.9, 1.5, and 2.0. It is further assumed that the respective airplane L/D's at these Mach numbers are 15, 6, and 3 and the respective engine efficiencies, with the afterburner operating, are 12 percent, 18 percent, and 23 percent. For $M_a = 0.9$, engine efficiency is assumed to be 22 percent if the afterburner is not operating. These specific L/D engine efficiency points are plotted and designated by the corresponding airplane Mach numbers. The several radii of action for the interceptor operating with afterburner at different Mach numbers are thus determined, and a curve connecting the points represents the airplane radius of action envelope. This curve is marked "current." The single point for the non-afterburner engine at an airplane speed of $M_a = 0.9$ is also plotted. An improvement in engine efficiency of 50 percent (essentially, a 50 percent improvement in the product $h\eta_e$) at all Mach numbers and of 50

percent in L/D at speeds at or above $M_a = 1.5$ is next assumed. At $M_a = 0.9$, the L/D is assumed to be 18 instead of 15. A new envelope curve, labeled "advanced," is drawn through these points. While no particular brief is held for the values shown, the curve marked "current" is reasonably representative of current practice, and the curve marked "advanced" is reasonably representative of future practice for which the necessary development information is being acquired. For a one-third increase or a one-third decrease in the cruise fuel percentages chosen
3460 (that is for a total fuel load between about 35 and 55 percent of the gross weight at take-off), the radii of action shown can be considered to vary directly with the fuel load available for cruise.

A discussion of the effect that fuel heat of combustion has on range should include nuclear-powered propulsion systems. For such systems, the number of miles $h\eta_e$ for which each pound of fuel burned will deliver one pound of thrust is many times the value of 240 to 1000 nautical miles given for current turbojet propulsion systems and fuels. In fact, the $h\eta_e$ value is sufficiently high that the weight of fuel is negligible; if a nuclear-powered airplane will fly at all, its range will be adequate. The efficiency of such an engine is important, however, from considera-tion of the thrust produced per pound of air consumed per unit of time. (It is assumed that heat produced by nuclear energy is being applied to an otherwise approximately conventional turbojet engine.)

The primary factor that will determine the success of a nuclear air-craft is, therefore, specific engine weight. The relationship of interest in regard to the application of nuclear energy is:

$$\frac{W_e}{F_{av}} \leqq \frac{W_e}{W_g} \cdot \frac{L}{D} \tag{6}$$

in which F_{av} is the thrust available at the flight condition. Equation (6) states simply that for horizontal flight at constant velocity, thrust available must be equal to or greater than airplane drag (thrust required). The ratio of power plant weight plus fuel weight to airplane gross weight $\frac{W_e + W_f}{W_{g_o}}$ can probably be about the same for the nuclear power plant as for the chemically fueled aircraft. In this comparison, $W_e + W_f$ for the nuclear power plant does not include the weight of fuel for a chemically powered supersonic dash if such is to be employed. Using a value of 0.65 as representative of the maximum permissible value for the ratio $\frac{W_e + W_f}{W_{g_o}}$, permissible specific power plant weight for the nuclear powered airplane becomes:

$$\frac{W_e}{F} \leqq 0.65 \frac{L}{D} \tag{7}$$

Equation (7) represents, then, the approximate relationship that must be satisfied. Because values of L/D for speeds in excess of $M_a = 1.5$ are less than approximately 1/3 those for subsonic speeds, it will be difficult to satisfy requirements of supersonic nuclear-powered flight. The difficulties are partially offset by the fact that, considering current turbine-inlet temperatures, at $M_a = 1.5$ and $M_a = 2.0$ specific engine weights are, respectively, 0.7 and 0.5 as much as the specific weight at $M_a = 0.9$.

Effects on Ratio of Military Load to Gross Weight 3460

For those conditions under which airplane gross weight is limited, means are desired that will permit military load to be increased without increasing gross weight or decreasing range. For those conditions in which range is of secondary importance, it may be desirable to decrease gross weight for a given military load. In either case it is desired to increase W_m/W_{g_o} without decreasing range. Equations (4) or (5) will therefore be examined for the condition of constant range to determine the extent to which W_m/W_{g_o} can be increased by varying each of the other five variables.

Figure 8 shows the results for a bomber and figure 9 for a fighter. For these curves each of the major variables is varied in turn in equations (4) or (5) with an accompanying variation in W_m/W_{g_o} so that the range remains constant. The abscissas of the curves are, as in figures 4 and 5, an increase in h, η_e, or L/D or a decrease in W_a/W_{g_o} or W_e/F_o. The ordinate is either W_m/W_{g_o} (which can be considered as a variation in W_m for constant W_{g_o}) or gross weight as indicated. Since there is some limit to how high the ratio W_m/W_{g_o} can go, for a given military load or for a given gross weight, the curves are not extended beyond W_m/W_{g_o} of 0.150, that is a gross weight of 50 percent of the original weight nor is consideration given to the extent to which it is practical to approach this 50 percent figure. Increase in h or η_e increases miles flown per pound of fuel burned, and therefore less fuel need be carried. The resulting numerical decrease permitted in W_{f_o}/W_{g_o} may then be added directly to W_m/W_{g_o}. Actually, referring to equation (4), $\log_e \frac{1}{\frac{W_a}{W_{g_o}} + \frac{W_m}{W_{g_o}} + \frac{W_e}{W_{g_o}}}$ is decreased to compensate for the increase in h or η_e, and this decrease is achieved through increasing W_m/W_{g_o}.

To determine the effect of change in L/D, the same limiting assumptions as were discussed for figures 4 and 5 were used. An increase in

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L/D permits a corresponding decrease in the logarithm in equation (4) with a consequent increase in W_m/W_{g_o}; and, for the case in which $\frac{W_e}{W_{g_o}} \cdot \frac{L}{D} =$ constant and specific engine weight is constant, there is a numerical decrease in W_e/W_{g_o}, which is also added to W_m/W_{g_o}.

To determine the effects of changes in W_{af}/W_{g_o} and in $(W_e/F_o)/(L/D)$ (assuming constant L/D) the amount either factor is decreased is added directly to W_m/W_{g_o} so that the sum of the two remains constant. For example, a 25 percent decrease (from 0.300 to 0.225) in W_{af}/W_{g_o} for the bomber increases W_m/W_{g_o} from 0.075 to 0.150 (a 100 percent increase).

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For the bomber or the fighter, to maintain constant range for constant military load and decreasing gross weight, an increase in L/D is again the most effective change because the two-fold advantage of an increase in L/D is realized. A decrease in specific engine weight is much the least effective means. For the fighter, a decrease in engine weight is of course much more effective than for the bomber, and here an increase in h or η_e is least effective. As in figures 4 and 5, if an increase in L/D is accompanied by no change in the value of W_e/W_{g_o}, the effect of the change in L/D is the same as that of a change in h or in η_e.

Comparison of figures 8 and 9 with figure 3 shows that decreasing gross weight of the bomber or fighter by 50 percent or increasing military load 100 percent (increasing W_m/W_{g_o} from 0.075 to 0.150) decreases the range potential by 25 or 30 percent, respectively, from what it would have been had W_m/W_{g_o} been maintained at the 7.5 percent value. For this reason, this procedure is not used except in those cases where airplane range must be sacrificed to reduce gross weight or where range is secondary to weight of military load carried.

Effects on Airplane Altitude and Speed

Maximum permissible airplane altitude (airplane celing*) and speed (considering the trimmed airplane in level flight) are determined by five factors: (1) lift-drag ratio of the trimmed airplane as a function of altitude and Mach number, (2) specific engine weight as a function of altitude and Mach number, (3) the ratio of engine weight to gross weight, (4) permissible engine pressure loading, and (5) maximum permissible ambient stagnation (total) temperature. These values determine a limiting Mach number-altitude envelope for the airplane.

*Airplane ceiling is defined as that altitude at which maximum available engine thrust equals drag of the trimmed airplane in level flight, at the flight speed under consideration.

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Engine Specific Weight, Airplane L/D Ratio, and

Ratio of Engine Weight to Gross Weight

The interrelation of the first four factors can be expressed by the relationship:

$$\frac{W_{eng}}{W_{g,TO}} \geqq \frac{W_{eng}}{F_{TO}} \cdot \frac{F_{TO}/F_{av}}{L/D} \cdot \frac{W_g}{W_{g,TO}} \qquad (8)$$ 3460

in which F_{av} is the thrust available under the flight condition and W_g is the gross weight of the airplane at the instant under consideration, and in which the subscript TO signifies the take-off weight or thrust. It is noted that the specific weight of the engine at altitude and flight speed is $\frac{W_{eng}}{F_{TO}} \cdot \frac{F_{TO}}{F_{av}}$.

As has been mentioned previously, with current combat military aircraft, the value of $\frac{W_{eng}}{W_{g,TO}} + \frac{W_{f,TO}}{W_{g,TO}}$ is approximately constant (about 0.65 for a bomber and 0.55 for a fighter). Under this condition if higher altitudes are to be obtained without decreasing range, the thrust available must be increased without increasing the ratio $W_{eng}/W_{g,TO}$. Therefore, either the specific weight of the engine at altitude and flight speed must be decreased or the lift-drag ratio of the airplane must be increased. A decrease in either of the two ratios which determine specific engine weight at the flight condition is of interest. The possibility of such decreases will be discussed in the section on improvements in engine performance.

The interrelation of the factors in equation (8) can also be expressed in a revision of equation (6):

$$\frac{L/D}{W_{eng}/F_{av}} \cdot \frac{W_{eng}}{W_g} \geqq 1 \qquad (6a)$$

The maximum permissible altitude, or airplane ceiling, at any constant flight speed is that altitude for which the left member of the equation equals 1. It becomes necessary, therefore, to examine the effects of airplane speed and altitude on airplane lift-drag ratio and on engine specific weight.

In figure 10, representative values of airplane lift-drag ratios as realized or as estimated by the manufacturers (August 1953) are shown for current supersonic interceptors or fighters. The band shown covers the lift-drag ratios for the F-100, F-101, F-102, F-104, F-105, F-9F-9, and XF8U-1. In each case, the L/D of the trimmed airplane in level flight was used. The airplane weights are from 15,000 to 44,000 pounds and the altitudes considered from 35,000 to 55,000 feet. In general, the high limit of L/D shown represents the higher altitude for these airplanes and the lower limit the lower altitude. Similar data are not available for bomber aircraft, but as mentioned previously, an L/D of 1.5 to 2.0 times the values given for the fighter will provide reasonably accurate estimates for analyses such as presented herein. Possible improvements in L/D will not be discussed in this report other than that previously considered in connection with figure 7.

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Figure 11 shows the effect of altitude and airplane Mach number on specific engine weight. (The L/D curve shown will be discussed later.) The external drag of the inlet diffuser is not considered in estimating these specific weights since it has been included in the drag (D) of the airplane. The specific engine weights indicated are for an afterburner engine with the afterburner operating at a temperature of 3040° F. The values are representative of current practice. The specific weight of an engine is normally given as the specific weight at sea-level static conditions with the afterburner inoperative (military rated thrust). For the engine assumed, this value is 0.460 pound per pound of thrust. The take-off specific weight of the engine is (since use of the afterburner increases thrust at take-off 50 percent) 0.310 pound per pound of thrust. The sea-level static specific weight of the corresponding nonafterburner engine is 0.368* pound per pound of thrust. As previously noted, these values are 80 percent of the weight included in the installed power plant of table I. The curves of figure 11 show the degree to which specific engine weight decreases with airplane speed and increases with airplane altitude. These specific weights at flight conditions are those used in determining airplane altitude and speed limits.

For a fighter, the assumption is now made that engine weight is, as indicated in table I, 20 percent of the take-off gross weight (that is, installed power plant, 25 percent). From figure 6, because of the fuel consumed before combat, the gross weight at combat is 0.84 of take-off gross weight. From the previous assumptions, the engine weight is therefore, at combat, 24 percent of the combat gross weight. A value of 25

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*At a Mach number of 0.9, the specific weights for the nonafterburner engine are about 1.5 times the afterburner engine values given in figure 11. Removing the afterburner is assumed to remove 20 percent of the engine weight.

percent is used. Absolute ceiling of the fighter (eq. (6a)) under these conditions will therefore be that altitude at which L/D of the trimmed airplane in level flight is 4 times the specific engine weight. Corresponding values of L/D for the fighter are included as ordinates in figure 11. These are the L/D's that must be available if the airplane is to fly at the speed and altitude chosen.

If at each airplane Mach number, L/D of the trimmed airplane as a function of altitude is known, the altitude at which the drag of the trimmed airplane is equal to thrust available can be determined. If for each airplane Mach number (for an altitude range of 45,000 to 57,000 feet) the approximate maximum value of L/D is as given in figure 10, the airplane L/D - Mach number curve may be drawn as indicated in figure 11. The values are somewhat arbitrarily extended to $M_a = 3.0$. In this case, the effect of altitude on the L/D of the trimmed airplane is neglected.

The L/D curve in figure 11 is an envelope curve in relation to the specific engine weight curves, in that at each Mach number the airplane ceiling is that altitude at which the specific engine weight curve for that altitude intersects or is tangent to the L/D curve. Within the limitations of the assumptions, figure 11 emphasizes the need of lift-drag ratios higher or specific engine weights lower than those now available if altitudes of 65,000 feet or greater are to be achieved in level flight.

The absolute ceiling envelope as a function of fighter airplane speed and altitude can now be determined using the data in figure 11. The curve expressing this relationship is plotted in figure 12 and is labeled $\frac{L/D}{W_{eng}/F} \cdot \frac{W_{eng}}{W_{g_o}} = 1$. The shape of this curve is determined by the relation of the L/D curve to the specific engine weight curve.

Maximum Permissible Engine Pressure Loading

The next step is to consider the limitation placed on flight speed and altitude by permissible engine pressure loading. Current engines are generally limited to about that pressure differential which occurs at $M_a = 1.0$ at sea level. For full adiabatic ram pressure and a $M_a = 1.0$ limit at sea level this pressure differential is:

$$\Delta p_{max} = (1.89\ P_{r,c} - 1)\ 14.7\ \mathrm{lb/in.}^2 \tag{9}$$

in which:

Δp_{max} maximum permissible pressure differential, lb/sq in.

$P_{r,c}$ engine compressor pressure ratio (between 7.0 and 12.0 for current turbojet engines)

Knowledge of the numerical value of Δp_{max} permits calculation of that
3460 Mach number at each altitude which results in a pressure differential across the engine equal to Δp_{max}. Representative values are given in table II.

TABLE II. - EFFECT OF ALTITUDE ON AIRPLANE MACH

NUMBER FOR CONSTANT ENGINE PRESSURE LOADING

Altitude, ft	Airplane Mach number, M_a
S. L.	1.0
10,000	1.4
20,000	1.6
30,000	1.9
40,000	2.2

In determining these values, corrections to the engine compressor pressure ratio resulting from inlet-temperature variations with altitude and with airplane Mach number have not been made, nor have corrections been made for the effect of stagnation temperature on the engine stress limit. For these and other reasons, the data represent an approximation and should be so considered. The data from table II are plotted in figure 12 and labeled $\Delta p = \Delta p_{max}$. This curve represents that part of the altitude-speed limiting envelope that is determined by engine strength. If the engine were required to operate at higher Mach numbers at low altitudes, the casing would be made thicker. This would give the required strength but would make the engine heavier and performance at higher altitude would suffer.

Maximum Permissible Total Temperature

As airplane speed is increased, the total or stagnation temperature increases because of adiabatic ram compression. This successively higher total temperature is approximately the engine inlet air temperature that the turbojet engine must withstand. At an airplane speed of $M_a = 2.0$, this temperature is 240° F (NACA standard day) at altitudes above 35,000 feet. In figure 12 this value is plotted as the limiting temperature.

The three limiting curves that establish the fighter airplane altitude-speed envelope are now determined, and a single curve labeled "generalized estimate" is faired from them. Again it is emphasized that these values are for the trimmed airplane in level flight.

In addition to the generalized estimate curve, there is included a cross-hatched area representative of the area included between the limiting curves specified by the manufacturers for the USAF fighters listed previously. These data show that the generalized estimate is reasonably representative of current practice.

The relative effects of the various factors that may be varied to increase the altitude-speed area included in the envelope are shown in figure 13. Had the L/D values for speeds above $M_a = 2.0$ (fig. 11) been assumed to remain essentially constant, the corresponding curves in figure 13 would, for airplane speeds in excess of $M_a = 2.0$, slope upward instead of slightly downward. The curves as shown are nevertheless representative of the increases in airplane speed and altitude that will accrue from improvements in the limiting values of the factors involved.

The curves of figure 13 indicating the portions of the altitude-speed envelope limitations determined by L/D, W_{eng}/F, and W_e/W_{g_o} show the marked increases in airplane ceiling that can result from changes in these values. For instance, if the amount of metal in the engine relative to the engine size is reduced 30 percent and the rate of air flow through the engine (that is, thrust) is increased 30 percent, specific engine weight is decreased to 0.54 of its current value with a consequent increase in ceiling of 15,000 feet. If this improvement is combined with a 30-percent increase in L/D (23-percent decrease in drag), the specific engine weight is decreased to 0.41, and the ceiling is increased by an additional 5000 feet.

The curves also indicate the improvement in ceiling achieved through increasing that part of the gross weight allotted to the engine (W_e/W_g). A 30-percent increase in this ratio would cause a 7500-foot increase in ceiling. This engine weight increase would also require that the fuel load at take-off be decreased from 30 percent of the gross weight to 22.5 percent. Reference to the fuel allotment tabulation of figure 6 shows that this decrease in allowable fuel weight would, if applied to the cruise portion of the flight, reduce the radius of action of the interceptor by half.

Because of the general flatness of the $\frac{L/D}{W_e/F_{av}} \cdot \frac{W_e}{W_g} = 1$ curves (fig. 13), the limitation to maximum permissible speed is the engine-pressure-loading limit at the lower altitudes and the limitation imposed by the

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total temperature at the higher altitudes. Solution of the problems imposed by this temperature limitation will require much effort. The airframe, the propulsion system, and the military load are all affected about equally. The rate at which these problems can be solved will determine the time at which these higher speeds can be achieved for other than short bursts.

Heat of combustion of the fuel and over-all efficiency of the engine do not directly affect either airplane ceiling or maximum speed. They
3460 may be considered to have an effect because they determine (eq. (5)) the percent that $\frac{W_e}{W_g} = \left(\frac{W_e/F_{av}}{L/D}\right)$ may be increased for a given increase in h or η_e with R remaining constant. In this case, as the heat of combustion h or over-all efficiency η_e is increased, the fuel carried is decreased sufficiently to maintain range constant, and the saving in fuel weight is allotted to an increase in engine weight and consequently in thrust available. Reference to equation (5) and to figures 8 and 9 shows that maintaining range and ratio of military load to gross weight constant, an increase of 20 percent in either heat of combustion of the fuel or engine efficiency permits a 15 percent increase in the ratio W_e/W_g for the fighter and 50 percent for the bomber. The corresponding ceiling increases are approximately 4000 feet for the fighter and 13,500 feet for the bomber. The corresponding permissible L/D decreases for constant altitude are 14 percent for the fighter and 33 percent for the bomber.

Treatment of the effect of a decrease in specific airframe weight, W_{af}/W_g, in the same manner shows that a given percentage decrease in airframe weight will permit for the fighter a percentage increase in engine weight of 1.5 times the given value and for the bomber 2.4 times that value. These increases will be accompanied by corresponding increases in thrust available and therefore in ceiling or in speed.

Engine Size

Engine size is generally expressed in terms of sea-level static military rated thrust. The engine size required for any particular airplane is dependent on the product of the maximum value of the ratio of thrust required to gross weight, and the ratio of the thrust at take-off to the thrust at this flight condition, and on the number of engines installed in the airplane.

Estimates of the required engine size, assuming for instance that the combat condition determines the size, can be made as follows. The thrust required at combat is determined by the combat values of gross weight, lift-drag ratio, and acceleration required for maneuvering. From the thrust required at combat, military rated sea-level static thrust can be determined from the engine specific weight-altitude-speed relationship, as presented in figure 11. From this thrust and the percentage of gross

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weight that is allotted to the engine, the required sea-level static thrust and sea-level specific engine weight is determined. Sample values are presented in table III.

TABLE III. - RELATION BETWEEN AIRPLANE COMBAT ALTITUDE, SPECIFIC ENGINE WEIGHT*, AND AIRPLANE GROSS WEIGHT. $M_a = 2.0$.

(a) Long range bomber				
$L/D = 5$; $W_{eng}/W_{g_{TO}} = 0.10$	$W_{g_C}/W_{g_{TO}} = 0.67$; $W_{eng}/W_{g_C} = 0.15$			
Combat altitude, ft	45,000	55,000	65,000	75,000
$F_{TO}/W_{g_{TO}}$	0.214	0.329	0.523	0.850
Required TO spec. eng. wt	.466	.304	.191	.118
Required mil. rated SLS sp. eng. wt	.699	.456	.286	.177
Take-off gross weight				
Mil. rated SLS thrust, lbs				
40,000	280,000	183,000	115,000	70,000
56,000	390,000	256,000	161,000	98,000
80,000	560,000	365,000	229,000	141,000
112,000	-------	516,000	322,000	197,000

(b) Fighter				
$L/D = 3.0$; $W_{eng}/W_{g_{TO}} = 0.20$	1.2 g turn	$W_{g_C}/W_{g_{TO}} = 0.833$; $W_{eng}/W_{g_C} = 0.25$		
Combat altitude, ft	45,000	55,000	65,000	75,000
$F_{TO}/W_{g_{TO}}$	0.64	1.00	1.57	2.55
Required TO spec. eng. wt	.312	.200	.127	.078
Required mil. rated SLS sp. eng. wt	.463	.300	.190	.117
Take-off gross weight				
Mil. rated SLS thrust, lbs				
10,000	24,000	15,000		
14,000	33,000	21,000		
20,000	47,000	30,000	19,000	
28,000	65,000	42,000	27,000	16,000

*Airplane speed and altitude affect specific engine weight as given in fig. 11.

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From the values of table III, the relation between total engine military rated thrust and airplane gross weight is determined. In the computations, it is assumed that afterburner engines are used and, as specified previously, that the ratio of take-off thrust F_{TO} to sea-level static military rated thrust is 1.5. That is, the ratio of take-off specific engine weight to military rated specific weight is 0.67.

In table III(a), the required sea-level static military rated specific engine weight for the combat altitude of 55,000 feet is representative of current engine design (see fig. 11). The table brings out again that specific engine weight must be reduced below current values if altitudes of 65,000 feet or higher are to be attained at the assumed combat-speed and lift-drag ratio with current airplane weight distributions. The table also indicates the need for engines of greater sea-level static thrust or for a greater number of engines as combat altitudes are increased.

In table III(b) for the fighter, the same essential points are illustrated as for the bomber. In this case the values at 45,000 feet are representative of the values given in figure 11. It is also shown, based on the assumed airframe and engine performance and the airplane weight distribution, that thrust-wise, vertical take-off becomes a matter of choice for fighter airplanes with which combat altitudes of the order of 60,000 feet or higher can be attained. It is well to point out that certain engine improvements that are discussed later will decrease the ratio between the sea-level static thrust and the thrust at altitude for speeds in excess of $M_a = 1.3$.

The need of large engines or of multi-engined fighters is illustrated for the higher combat altitudes. For instance, at 65,000 feet combat altitude a 27,000 pound gross weight (at take-off) fighter requires a single engine of 28,000 pounds military rated thrust. The thrust available for take-off with the afterburner operating on such an engine is 42,000 pounds. Improvements in airplane combat lift-drag ratio or realization of the engine improvements mentioned above would do much to relieve this situation.

In table III, the values of military rated thrust listed are separated by increments of about 40 percent. If a maximum of eight engines is to be used in the largest long-range bombers, these data indicate that a relatively small number of different engines will provide for a wide range in gross weight for both fighters and bombers. Table III does not represent the complete relation of required engine sea-level static thrust to airplane gross weight, since the lift-drag ratio for only one altitude and flight speed is considered, but it shows the general effect of increased combat altitudes. A more complete evaluation should include a consideration of the acceleration and rate-of-climb requirements particularly in the transonic speed region.

Summary

The interrelationships between gross weight, airplane range, ceiling, and speed; and airplane weight distribution, airframe lift-drag ratio;

and propulsion-system characteristics have been examined. The detail of the examination has varied with the parameter. Because of contributing factors that have not been considered, the analysis is not intended to give precise values. It does provide a background for general research emphasis and for estimates in regard to the probable airplane performance that can be obtained through improvements in the variables considered. Of the information presented, tables I and III, equations (3), (5), and (8), and figures 1, 3, and 13 are probably of most significance.

Gross weight of the airplane is largely a function of military load (fig. 3) and, unless there are considerations limiting gross weight or required range, gross weight of the airplane will be of the order of 13 times the weight of the military load. Figure 1 and equations (3) and (5) show the manner in which the primary airplane variables affect range. Increases in range will most likely be realized through increases in heat of combustion of the fuel, in propulsion-system efficiency, and in lift-drag ratio of the airplane. Appreciable decreases in specific airframe weight do not appear likely. Increases below current values in the ratio of military load to gross weight are not an effective means of increasing range (fig. 3). A decrease in specific engine weight will most likely be used to increase permissible airplane speed and altitude rather than to increase range. Since increases in range resulting from increases in airplane lift-drag ratio, engine efficiency, and fuel heat of combustion are cumulative, small increases in each are well worth while. 3460

Figure 13 and equation (8) show approximate limits of current fighter-airplane speed and altitude. To the extent that the engine in the fighter or bomber will constitute a fixed percentage of the gross weight, increased altitude and speed will be obtained jointly through decrease in specific engine weight and increase in airplane L/D, both improvements giving quantitatively about the same results. A 10-percent improvement in either will increase airplane ceiling by 2500 or 3000 feet. As mentioned previously, the increase in L/D ratio will also increase range.

The rate at which maximum airplane speed can be raised to values above $M_a = 2.0$ will depend largely on the rate at which successively higher stagnation temperatures can be tolerated and (at lower altitudes) the rate at which maximum permissible engine pressure loads can be raised.

Engine size (thrust wise) is largely a function of airplane ceiling, maximum speed, and number of engines per airplane. As the ceiling is raised, larger engines or a greater number of engines will be required.

The next step in the present analysis is to estimate the extent to which each propulsion-system factor can be improved, considering both degree of improvement and possibility of achieving the improvements.

The discussion will cover improvements in specific engine weight, engine efficiency, maximum permissible engine pressure loads, and fuel heat of combustion. Following these discussions the fuel availability, critical materials, and lubrication problems will be discussed.

CURRENT STATUS AND ESTIMATED IMPROVEMENTS

IN PROPULSION SYSTEM PERFORMANCE

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The propulsion variables under consideration will be taken up in order: specific engine weight W_{eng}/F, engine efficiency η_e, and fuel heat of combustion h. For each variable, an estimate is made of the present position and of the possibilities for improvement.

Figure 14 is a diagrammatic sketch of a supersonic turbojet engine. The combined aerodynamic and thermodynamic performance of the six major engine parts indicated, together with the characteristics of the materials from which the engine is made and of the fuel from which the propulsive energy is derived determine specific engine weight and engine efficiency for any flight condition.

Certain factors that affect specific engine weight also affect engine efficiency. In general, the relation of these factors to specific engine weight is discussed in the section on specific weight, and the relation to engine efficiencies in the section on efficiencies.

Specific Engine Weight

As mentioned previously, the specific weight guaranteed by the engine manufacturer is the military rated specific weight, that is, the specific weight under sea-level static conditions at rated engine speed with the afterburner not operating. The general effect of airplane altitude and speed on specific engine weight at different flight conditions has been presented in figure 11. Under any operating condition, specific weight of the engine is established for the most part by three variables:

1. Weight of material in the engine

2. Rate of air flow through the engine

3. Temperature to which the air is burned, which in turn determines the thrust produced per pound of air flowing through the engine.

Weight of material in the engine is, of course, a constant for a given engine. Rate of air flow through the engine is dependent first on

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the compressor design and secondly, considering full-throttle operation, on the altitude and speed at which the airplane is flying. For airplane speeds up to $M_a = 1.0$ at sea level and up to $M_a = 1.3$ at altitudes of 35,000 feet or over, the predominant effect of airplane speed and altitude is change in density of the air at the compressor inlet. For these flight conditions, air-flow rate through the engine can be considered as the product of the air-flow rate under the sea level static condition and ρ/ρ_{sl}, in which ρ_{sl} is the density of the air under sea level static conditions and ρ is the density of the air at the compressor inlet under the specified flight condition. At airplane speeds higher than $M_a = 1.0$ at sea level or $M_a = 1.3$ above 35,000 feet, rate of air flow through the engine is further affected by changes in inlet-air temperature and velocity, because they alter aerodynamic performance of the compressor and aerodynamic and thermodynamic performance of the inlet diffuser and exhaust nozzle. The effect of these factors on rate of air flow and on thrust produced, and thus on specific engine weight, will be discussed in the order just given.

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The effects on specific engine weight of combustion temperatures, whether in the combustor or in the afterburner, and the factors that limit these temperatures are referred to briefly in this section on specific engine weights. They are discussed in more detail in the section on engine efficiencies, since their effect on specific weights is closely interrelated with their effect on efficiencies. The use of special fuels to attain higher combustion temperatures is discussed in the section on fuels.

Weight of Materials in the Engine

An analysis has been made by James Lazar (at the time, of the NACA Headquarters staff) of engine weights guaranteed or estimated by manufacturers (as of August 1953). These data represent all U. S. axial-flow turbojet engines having military rated thrust of 3000 pounds or higher from the J-34 through the J-79 engine. The analysis shows that a reasonably good correlation is obtained if engine weight is plotted against the cube of engine compressor tip diameter, figure 15. As mentioned previously, for a given compressor tip diameter an engine without afterburner has approximately 80 percent the weight of an afterburner engine.

The engines are divided into three groups according to their stage of development as of approximately August, 1953. Those engines that had passed the 150-hour test are represented in figure 15 by diamonds "◆", those that were in the process of passing this test by squares "■", and those that were in the design stage or the early hardware stage by circles, "●". With one exception, the weights of engines in the design and early

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hardware stages are described quite accurately by the straight line of equation (10). One of the points on the curve of figure 15 is an Americanized engine of British design. The data of figure 15 show the present state of the art and the reasonable uniformity of engine-weight patterns being followed by different engine manufacturers.

In the development of these engines, emphasis has been placed on obtaining higher thrusts and higher pressure ratios without increasing specific engine weight. Considerable emphasis is now being placed on reducing the amount of metal in an engine of given diameter, to decrease specific engine weight. Consequently, figure 15 and equation (10) should not be used to estimate future engine weights.

The relation of various engine diameters to compressor tip diameter are as shown in table IV.

TABLE IV. - RATIO OF OUTSIDE DIAMETER OF SEVERAL ENGINE PARTS TO COMPRESSOR TIP DIAMETER

	Production engines	Development engines
Compressor case	1.16	1.07
Combustor	1.24	1.08
Turbine	1.23	1.05
Afterburner	1.32	1.17
Engine envelope circle	1.53	1.35

In table IV, the "production engines" in general represent those engines indicated by diamonds in figure 15 and the "development engines" represent those engines indicated by squares and circles.

Rate of Air Flow Through the Engine

The amount of air flowing through the engine is the second of the factors controlling engine weight to be considered. At any inlet condition the air flow per square foot of compressor frontal area, based on the tip diameter, figure 16, is a function of the average air velocity into the compressor and of the ratio of the compressor hub diameter to tip diameter (hub-tip ratio). An increase in this velocity or a decrease in hub-tip ratio increases this specific air flow. The data for figure 16 are based on the average inlet axial Mach number for the annular flow area immediately ahead of the first-stage compressor tip diameter. Standard sea-level conditions are assumed. The maximum air flow rate shown, 50 pounds per square foot per second, is the value for a "choked" pipe under standard sea-level conditions.

[redacted]

The shaded area marked "subsonic" is representative of present practice. The turbojet engines that were in the process of passing or had just passed the 150-hour test (as of August 1953) have an air flow of about 25 pounds per second per square foot. At this air flow, for a 35-inch diameter compressor tip and 62.5 pounds of thrust per pound of air per second, the military rated sea-level static specific engine weight is 0.460 (fig. 15). The newer engines in the design and early hardware stage have an air flow of 27 to 30 pounds per second per square foot. The major axis of the shaded oval area (fig. 16) marks the general progress to these higher air-flow values by the use of higher inlet velocities and lower hub-tip ratios. 3460 Values of military rated specific engine weight as a function of thrust for the afterburner engine are shown in figure 17, assuming the weight-diameter relationship given in figure 15 and assuming 62.5 pounds of thrust per pound of air flow per second. The curve is representative of current engines and should not be used to estimate future specific engine weight.

As inlet Mach number is increased, a point is reached (if the first compressor stages are to develop reasonably high pressure ratio) at which flow velocity relative to the compressor blade tips exceeds a Mach number of 1.0. This condition requires that the air foil design of compressor blades in the stages involved be transonic instead of subsonic. Current research and development data indicate, as denoted by the area marked "transonic", that by using a transonic compressor having a hub-tip ratio of 0.35 to 0.40, air-flow rates approaching 40 pounds per second per square foot are feasible. At the same time, the pressure ratio developed by the transonic stages will exceed that developed by their subsonic counterparts. Application of transonic compressors having one or more transonic stages is being fostered by the NACA and the engine industry.

Blades having transonic air foils have operated successfully with relative Mach numbers of 1.2 at the tip in a research multistage compressor. An experimental stage, operated at a tip Mach number of 1.4, indicated high efficiency. At this Mach number level, the air foils might be more properly described as supersonic. There is some promise that efficient supersonic air foils will be used for compressor blades in the future. Such air foils will, however, probably be exploited for increasing pressure ratio of the stages involved and thereby decreasing engine weight, rather than for increasing compressor air flow capacity.

Consideration of higher rates of air flow through the compressor should include an examination of the ability of other engine components (fig. 14), notably the combustor, the turbine, and the afterburner, to handle increased rates of air flow. With current production engines, see table IV, the outside combustor diameter is about $1\frac{1}{4}$ times the compressor tip diameter. Consequently, at sea-level static conditions, for the production engines, an air flow, W_{ar}, of 25 pounds per second per square foot through the compressor corresponds to 16 pounds per second per square foot through the circle described by combustor outside diameter.

In the newer engines, the ratio of combustor outer diameter to compressor tip diameter is reduced to 1.10 or slightly less. For these engines, the compressor air flow of 30 pounds per second per square foot corresponds to a combustor air flow of about 25 pounds per second per square foot of frontal area and the compressor value of 40 pounds per second per square foot corresponds to a combustor value of 33 pounds per second per square foot. These higher rates of air flow through the combustor generally tend to increase combustor pressure drop, which results in a greater relative loss in thrust. Higher rates of air flow also tend to decrease combustion efficiency. These two losses can be combined into a combustor efficiency which expresses them as a reduction in engine efficiency.

Ability of the combustor to handle the higher air flow rates can be expressed as an airplane Mach number - altitude curve for a given percent combustor efficiency. Such a plot is shown in figure 18 for a combustor efficiency of 95 percent. For comparison, the current fighter limiting Mach number - altitude envelope from figure 12 is included. The combustor data plotted represent values obtained at the Lewis laboratory (ref. 1) with an experimental annular combustor having an inner diameter 0.4 of the outer diameter.

For an engine with a compressor pressure ratio of 7, the maximum altitude at which 95-percent combustion efficiency is obtainable is appreciably higher than current fighter altitude ceilings. Increasing sea-level static pressure ratio to 12 increases combustor altitude limits by providing less severe conditions for combustion and reducing the effect of compressor pressure drops on engine thrust. The data indicate that performance of the combustor need not delay the use of the higher engine air flow rates under discussion.

The problems of handling greater air flow through the afterburner are similar to those of the combustor. Afterburner altitude limits, based on data obtained with an experimental afterburner at the Lewis laboratory (ref. 2), are shown in figure 18 for an afterburner efficiency of 85 percent. (The curve for an 80 percent efficiency would be about 6500 feet higher.) Compressor pressure ratio has little effect on the altitude limits imposed by afterburner combustion efficiency. The data show that improvement in afterburner altitude efficiency is required to use effectively the higher air flow rates through the engine.

The problem of handling higher rates of air flow through the turbine has been discussed in reference 3 by Cavicchi and English.

To increase rate of air flow through the turbine, either turbine-outlet area must be increased or the product of density of the gas and axial velocity of the gas at the turbine outlet must be increased. If the area is increased and the ratio of compressor to turbine diameter maintained constant, the turbine hub-tip ratio must be decreased with consequent increase i[redacted] increase will be closely proportional to the increase in gas flow rate. Since the turbine outlet is close to the choked condition, to raise the outlet velocity, the turbine-outlet temperature, and therefore the turbine-inlet temperature must be raised. Considering the interrelationships of the different variables involved, the flow will be dependent on turbine-inlet temperature and compressor pressure ratio in the manner shown in figure 19 for a flight Mach number of 1.8 and a turbine-exit axial Mach number of 0.7. Since choking occurs in the rotor passage for an average exit axial Mach number near 0.7, the air flows given in figure 19 approximate the limiting values. The effect of hub-tip ratio is also shown in the figure. The curve for a turbine-inlet temperature of 1540° F and a turbine hub-tip ratio of 0.65 corresponds to present practice. The data show that for air flows in excess of 30 pounds per square foot, either increased hub-tip ratio or higher turbine-inlet temperature must be employed.

According to the data in table XIV the flow rate through the turbine should be about the same as that through the compressor. At current turbine-inlet temperatures and constant hub-tip ratio (constant annular area) increasing compressor pressure ratio has little effect on flow capacity because the drop in pressure across the turbine is almost as great as the rise in pressure across the compressor. At the higher compressor pressure ratios, however, large increases in flow capacity of the turbine can be obtained by increasing turbine-inlet temperature. For example, at a compressor pressure ratio of 12 an increase in turbine-inlet temperature from 1540° to 2540° F increases flow capacity approximately 30 percent. Or a decrease in the hub-tip ratio from 0.65 to 0.50 permits the turbine air flow to be increased to approximately the values required to take full advantage of the compressor air flow increases discussed in relation to figure 16. For no change in engine speed, this decrease in turbine hub-tip radius ratio, however, is accompanied by a 33-percent increase in blade stresses.

The amount that turbine air flow capacity can be increased without an accompanying increase in engine weight is thus dependent on the amount that increases in turbine-inlet temperature, or turbine blade stress, or both can be tolerated. With present day turbine materials, these requirements are conflicting.

With current turbine-inlet temperatures and turbine materials, the turbine-blade tensile stresses at the critical blade section are (expressed as 1000-hour rupture values) limited to about 25,000 pounds per square inch. Figure 20 shows an approximation of the effect of turbine-blade temperature and material on the 1000-hour rupture stress. The data shown by the solid lines were supplied by Mr. William L. Badger of the General Electric Company. The dashed lines represent some newer alloys that are in the early development stage. The curves show that the current cobalt- or nickel-based alloys are satisfactory stresswise for the present turbine-blade temperatures of about 1500° F, but that for higher stresses the blades must be cooled to lower temperatures. To permit stresses 50 percent higher than current values, the blade temperatures must be decreased 150° F. For twice the current permissible stress, the required decrease is 225° F. The nickel- and cobalt-base alloys under development may of themselves extend the permissible stress to 30,000 or 35,000 pounds per square inch, an increase of about 30 percent. The molybdenum-base alloys would provide still greater strength but until the problem of oxidation of the alloy is solved, its satisfactory use cannot be assured. Considering current blade material and turbine-inlet temperatures, the required increase in blade stress can be obtained by air-cooling the turbine wheel and blades. Figure 21 indicates the amount of engine air that must be bled from the compressor outlet to provide this cooling. These data apply to the blade type shown in figure 21. The loss in the engine efficiency with turbine cooling is negligible. The loss in thrust as will be discussed later is about equal to the percent of cooling air required, about 2 percent for a 300° F temperature drop for the blade shown. The possibility of increasing the turbine-inlet temperature through turbine cooling will be discussed in the section on engine efficiencies.

Effect on Engine Specific Weight of Aerodynamic and Thermodynamic Performance of Inlet Diffuser, Compressor, and Exhaust Nozzle

Decrease in the weight of material in the engine, as has been mentioned, results in a proportional decrease in specific engine weight regardless of flight condition. The same relation is true for the increases in rate of air flow through the engine so far discussed. In addition, this increase in air flow decreases engine frontal area for a given thrust. Those situations will now be considered in which, at high flight speeds, research and development indicates that the rate of air flow through the engine or the thrust produced per pound of air flow can be increased by further improvements in the compressor, the variable-inlet diffuser, and the discharge nozzle. Improvement of the compressor will place increased demands on the turbine. Means of meeting these demands will be discussed.

The airplane speeds at which such effects become appreciable are, as has been mentioned previously, in excess of $M_a = 1.0$ at sea level and in excess of $M_a = 1.3$ at altitudes of 35,000 feet or greater. Changes that may effect improvements at these speeds must be considered to determine if they will cause any loss in performance at the lower speeds. The engine compressor and the related turbine problems will be discussed first.

Compressor

The improvements in air-handling capacity of the compressor that have been discussed so far are a function of what has been termed aerodynamic design of the compressor. This air-handling capacity is a function of the annular flow area at the first compressor stage and (neglecting Reynolds number effects) of the permissible average inlet Mach number, relative to the peripheral area at this stage. As airplane speed is increased, the total temperature of the air is increased by ram compression of the air in the inlet diffuser. The effect of this temperature increase on compressor performance will be discussed in this section. 3460

If Reynolds number effects are neglected, the aerodynamic performance of any row of blades in the compressor, and therefore the rate of air flow into the compressor, depends on two factors: Mach number of the entering air, relative to the blades, and angle of attack of the blades, relative to this air.

The resultant Mach number and angle of attack are functions of the peripheral Mach number M_{per} of the compressor blades and the axial Mach number M_{ax} of the entering air as shown schematically in the diagram of figure 22.

Peripheral Mach number M_{per}, is given by:

$$M_{per} = \frac{\pi d_c N}{\sqrt{\gamma R_{ar} t_2}} \tag{11}$$

in which

d_c compressor diameter at blade section of interest

N compressor revolutions per unit time

R_{ar} gas constant for air

γ ratio of specific heats

t_2 static temperature of the entering air

Inlet axial Mach number M_{ax} is mainly a function of the peripheral Mach number:

$$M_{ax} = f(M_{per}) \tag{12}$$

For a particular engine, at constant values of turbine area and engine exhaust nozzle area, there is one and only one average axial Mach number for each peripheral Mach number. A curve representing this relationship is given in figure 22. The coordinate values above the abscissa axis and to the right of the ordinate axis will be discussed later. Since each point on the curve represents a unique value of Mach number and angle of attack relative to the compressor blades, each point also represents a unique value of pressure ratio, of compressor efficiency, and of inlet axial Mach number. Values of pressure ratio relative to the pressure ratio at the design peripheral Mach number and values of compressor efficiency are indicated on the curve. For this example, the design peripheral Mach number is 0.96; on the abscissa scale, this value is labeled "rated."

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The upper end of the curve has two limits. The first is a mechanical limit imposed by the maximum permissible engine stresses produced by rotative speed. The second is an aerodynamic limit and is that peripheral Mach number that results in compressor choking or in other aerodynamic limitations such as the need for blade matching between stages. Equation (11) shows that this aerodynamic limit may be reached either by increasing the engine rotative speed or by decreasing the temperature of the air at the compressor inlet.

The compressor is generally so designed that under some subsonic flight condition, the maximum permissible peripheral Mach number occurs at the maximum permissible speed. In the figure it is assumed that the maximum peripheral Mach number occurs at the maximum permissible speed, at a flight speed of $M_a = 0.6$, and an altitude of 35,000 feet or more. Rated speed and rated peripheral Mach number are given as 90 percent of these values.

The rate of air flow w_{ar} into the compressor is given by:

$$\frac{w_{ar}}{A_c} = \frac{A_n}{A_c} \rho v_{ax} \tag{13}$$

in which

A_c compressor frontal area based on compressor tip diameter

A_n annular flow area at first compressor stage

ρ density of inlet air

v_{ax} axial velocity of inlet air

Since:

$$v_{ax} = M_{ax}\sqrt{\gamma R_{ar} t_2} \tag{14}$$

in which

t_2 static temperature of the entering air

equation (13) can be rewritten as: 3460

$$\frac{w_{ar}}{A_c} = \frac{\rho}{R_{ar} t_2} M_{ax}\sqrt{\gamma R_{ar} t_2} \cdot \frac{A_n}{A} \tag{15}$$

Combining the temperature terms and converting to total temperature and pressure at the compressor inlet instead of static pressure and temperature, equation (15) becomes:

$$\frac{w_{ar}}{A_c} = \frac{A_n}{A_c} P_2 \sqrt{\frac{\gamma}{RT_2}} \left(1 + \frac{\gamma-1}{2} M_{ax}{}^2\right)^{\frac{\gamma+1}{2(1-\gamma)}} \tag{15a}$$

Equation (15a) shows the manner in which the rate of air flow into the compressor varies with the total temperature and total pressure and with the axial Mach number. The relation between axial Mach number and peripheral Mach number is shown in figure 22. The values in figure 16 were determined from equation (15a) for conditions of standard sea level total pressure and temperature.

If the engine is operating at rated rpm but the airplane is flying at such a speed that compressor inlet-air temperature, because of ram compression, exceeds sea-level standard air temperature, 60° F, the peripheral Mach number will be decreased below the rated value. Values of compressor inlet-air static temperature, corresponding to airplane Mach number from 0.6 to 3.0 at altitudes of above 35,000 feet are given together with the airplane Mach number on the abscissa in figure 22. The values are tabulated at the corresponding peripheral Mach numbers, equation (11), for each respective temperature with the compressor operating at rated rpm. The airplane Mach numbers are also tabulated on the ordinate axis at the corresponding axial Mach numbers. Although the rpm of the compressor has been maintained at its rated value, the air flow at flight speeds in excess of $M_a = 1.3$ has been decreased below that which the aerodynamic speed limit of the compressor would permit. The amounts are determined from figures 22 and 16 and are tabulated in table V.

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TABLE V. - LOSS IN COMPRESSOR PERIPHERAL MACH

NUMBER AND AIR FLOW AT RATED COMPRESSOR RPM

[Altitude greater than 35,000 ft]

Airplane speed, M_a	Percent rated, M_{per}	Loss in air flow, percent
2.0	85	22
2.5	75	37
3.0	68	48

The specific engine weights given in figure 11 include these losses.

If engine rpm could be increased to values of 116 percent of rated speed at $M_a = 2.0$, 130 percent at $M_a = 2.5$, and 147 percent at $M_a = 3.0$, air flow would in each case be increased to the "rated flow" and specific engine weight decreased by the respective amount shown in the loss-in-air-flow column. These increases in rpm would increase turbine blade or compressor stresses (which vary as the square of engine speed) 35 percent, 69 percent, and 116 percent, respectively. If the data in the table were based on the maximum permissible M_{per} of 1.07 M rather than the rated value of 0.96 M the air flow gains would be accordingly greater.

With a constant exhaust-nozzle area, increasing peripheral Mach number to the rated value also increases compressor pressure ratio to the rated value. This combination of high compressor-inlet air temperature and high compressor pressure ratio results in compressor-outlet temperatures higher than desirable. Increasing exhaust-nozzle area at the same time that engine rpm is increased, will result in a lesser increase in compressor pressure ratio than shown in figure 22.

The work of Cavicchi and English, previously referred to, has shown that the increase in turbine stresses accompanying the higher engine speeds will be one of the most difficult problems to overcome in bringing the compressor to rated aerodynamic performance at high flight speeds. The discussion on turbine stresses in relation to figures 19 and 20 applies. The cooling required to permit the higher stresses discussed here will be in addition to that required for this previous use. For instance, to decrease hub-tip ratio from 0.65 to 0.50 and at the same time to permit engine speed to be increased 16 percent (see discussion of table V and fig. 19) would require a stress increase (1.43×1.35) between 90 and 100 percent.

Increasing engine rpm for high flight speeds without increasing engine weight also poses problems such as the ability of the compressor to withstand the higher stresses. Adequate solutions cannot be assured. However, it should be remembered that the procedure discussed here provides a means for increasing airplane ceiling by 6000 at a speed of $M_a = 2.0$ and by 9000 feet at $M_a = 2.5$. 3460

Inlet Diffuser

The air-inlet and diffuser system plays an important role in the determination of effective engine specific weight. Mass flow through the engine, and consequently thrust, is directly proportional to the intake-system pressure recovery. In addition, thrust per pound of air is a function of the pressure recovery, through its effect on over-all engine pressure ratio. Drag produced by the intake system may also be viewed as a thrust decrement. Therefore, for minimum effective engine specific weight in a given aircraft configuration, inlet pressure recovery should be at the highest possible value throughout the flight range and inlet drag should be a minimum.

These inlet requirements can generally be met satisfactorily in subsonic airplanes through the use of fixed-geometry intake systems, inasmuch as such inlets can be efficient over a wide range of air flow and flight-speed conditions. At supersonic speeds, however, inlets have only a narrow operating range of flight speed over which they can deliver air at high pressure recovery and low drag. When flight speed is changed, it is generally necessary to vary the inlet geometry in order to obtain both high pressure recovery and low drag.

As maximum supersonic speed of the airplane is increased, the limitations of a fixed-inlet design become increasingly severe. It therefore becomes increasingly necessary to provide an air-inlet and diffuser arrangement that can be varied both in area and in relative shape as flight Mach number is changed. If this requirement is not met, there will be serious thrust losses during some portion of the flight speed range.

The extent of the thrust losses that may be incurred with fixed-geometry intake systems and the gains to be obtained from the use of variable-inlet geometries are illustrated in figure 23. These curves are based on calculated inlet performance data that are in good agreement with experimental values. Although the numerical results presented in the figure are for an engine operating at constant mechanical speed, the inlet-matching problem is qualitatively the same for engines employing increasing mechanical speed with increasing flight Mach number.

Curves are shown for inlets having design Mach numbers of 0.85 and 2.0. The design Mach number indicates the flight speed at which the inlet would supply the engine air flow requirements without spillage (low drag) and at the maximum inlet pressure recovery currently possible with moderate geometric complication. Both inlet designs incorporate comparable compression surfaces so that they would be capable of operating at the same pressure recovery levels through the flight speed range. (An inlet designed for $M_a = 0.85$ without a compression range would achieve the same subsonic performance levels as those shown in figure 24.
3460 At supersonic speeds, however, the thrust would be greatly reduced from the indicated values because of the low pressure recoveries associated with simple normal-shock inlets.) At other flight speeds, the inlets would either operate with reduced pressure recovery, or with spillage and increased drag, depending on the inlet design Mach number. The increased drag, where present, has been included in the net thrust evaluation shown in the figure.

The performance attainable with a variable-geometry inlet is also indicated in figure 23. This inlet is presumed to incorporate a variable-angle compression surface. The angle is reduced as flight speed is reduced along such a schedule that pressure recovery is maximized at each flight speed, while air spillage is reduced to a low value. Similar, though not identical, performance could be attained with inlets in which the compression surface is retracted as Mach number is reduced, or in which excess air is discharged through low-drag bypass ports ahead of the compressor face.

Using the variable-inlet performance as a standard, each of the fixed geometry inlets shows serious deficiencies. As a result of high air-flow spillages, the inlet having a design Mach number of 0.85 suffers a 16 percent loss in available thrust at a Mach number of 1.5. This loss is reduced to 5 percent at a Mach number of 2.0. The fixed-geometry inlet having a design Mach number of 2.0, on the other hand, eliminates the thrust loss at Mach number of 2.0, but incurs losses as great as 22 percent of the variable-inlet values at Mach number 0.85. These losses are caused by low operating pressure recoveries. Airplane operation to higher supersonic speeds than those shown on figure 24 will increase the penalties of fixed-inlet operation.

In the computations for figure 11, which shows the effect of airplane speed and altitude on specific engine weight, diffuser performance similar to that of the variable inlet was assumed.

Exhaust Nozzle

The exhaust-gas nozzle should provide a throat area corresponding to specified conditions of temperature, pressure, and weight flow at the

[redacted]

afterburner outlet. Failure to provide the proper area forces an adjustment in upstream conditions with a concomitant change in engine operation and a loss in thrust. This loss in thrust results from a loss in nozzle efficiency that reduces over-all engine efficiency. It can also cause throttling of the air flow through the engine, with losses of the type discussed in connection with the diffuser. To fulfill the requirements, throat area of the exhaust nozzle must be much larger for the afterburning engine than for the nonafterburning engine; smaller variations are required by changes in flight altitude and flight Mach number. Maximum jet thrust ideally occurs when the flow undergoes a controlled expansion to the condition where the jet static pressure equals the ambient pressure. A controlled expansion refers to one which takes place over thrust producing surfaces as contrasted with a free-jet expansion such as occurs with a simple convergent nozzle at pressure ratios higher than the critical choking value. The amount of expansion required increases with jet pressure ratio and hence with flight Mach number. 3460

The importance of properly expanding the flow at supersonic Mach numbers is illustrated in figure 24 (refs. 4 to 8) for an engine of advanced design operating at a constant rotative speed and with afterburning. The altitudes assumed are sea level for flight speeds below $M_a = 0.9$, and at the tropopause for flight speeds above $M_a = 0.9$. Four nozzle types are shown: (1) a convergent-divergent nozzle with variable throat and exit areas, (2) a plug-type nozzle with variable throat area that provides for controlled expansion over what might be considered an external surface, (3) an ejector nozzle with variable throat and shroud areas that expands the primary flow into a cushion of secondary air and thus provides thrust gains over the convergent nozzle by the mechanism of maintaining back pressure in the secondary flow passage well above ambient pressure, and (4) a simple variable-area convergent nozzle. The convergent nozzle becomes increasingly poor as flight speed increases until at $M_a = 2.5$ net propulsive thrust is only 77 percent of that ideally available. The convergent-divergent nozzle indicates less than a 5 percent thrust loss at the same flight speed. Unfortunately, the mechanical problem of varying the axially symmetric convergent-divergent nozzle geometry as required to obtain the indicated performance over the Mach number range is difficult. The ejector nozzle and the plug nozzle, however, represent practical configurations that can approximate the best convergent-divergent nozzle performance with realistic geometric variations. Cooling requirements have not been accounted for in this figure; only the ejector has potentially a built-in cooling system. In the calculations of engine or airplane performance presented herein, nozzle performance approximating that of the convergent-divergent nozzle was used.

Research and development are in progress both on the variable inlet and on the variable convergent-divergent nozzle. There is no reason to believe that satisfactory [redacted]

[redacted]

Combustion Temperatures

Specific engine weight under any flight condition can be further reduced by burning to higher turbine-inlet temperatures than the current maximum of about 1600° F. Since this effect is intimately related to the engine efficiencies, it is discussed later, in the section on efficiencies.

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Specific weight of the afterburner engine can be decreased if fuels can be used that will give higher afterburner combustion temperatures than those produced with hydrocarbon fuels. This possibility is discussed in the section on fuels.

Engine Efficiencies

Turbojet-engine efficiency is dependent on the efficiencies of the engine components; on engine compressor pressure ratio; on the temperature to which fuel is burned in the combustor and in the afterburner, if an afterburner is used; and on airplane speed and altitude. The effect of airplane altitude is secondary and is not considered here. Efficiency of the engine can be divided into two major factors: thermal efficiency and propulsive efficiency.

Thermal Efficiency

Thermal efficiency can be expressed as the ratio of the increase in kinetic energy of the gas through the engine to the chemical energy in the fuel used. Thermal efficiency is a function of: (1) over-all pressure ratio of the engine, which, in turn, is the product of ram pressure ratio in the engine diffuser resulting from the airplane velocity and engine compressor pressure ratio; (2) temperature to which the air is heated during combustion; (3) whether or not an afterburner is used; and (4) efficiency of the engine components. Thermal efficiency is measured by recording the data necessary to evaluate the equation:

$$\eta_{th} = \frac{1/2\ w_{ar} v_j^2 - 1/2\ w_{ar} v_a^2}{gJhw_a(f/a)} \tag{16}$$

in which

η_{th} thermal efficiency

w_{ar} mass rate of air flow through the engine

v_j jet velocity relative to the airplane

[redacted]

[redacted]

v_a airplane velocity

J mechanical equivalent of heat

f/a fuel-air ratio

Equation (16) neglects the effect of fuel weight.

Component Efficiencies 3460

The effect on thermal efficiency of certain of the component efficiencies is shown in figure 25, in which the efficiency of each component is given relative to its design point value, listed as the current value. The effects of diffuser and nozzle performance at supersonic speeds are represented only to a limited degree in this figure. In general, it can be said that improvements in the efficiency of any one engine component over the maximum values now obtained will not have a major effect on engine thermal efficiency, but since the effects of such improvements are complementary, small improvements in each are worth while. Examination of portions of the curves to the left of the "current values" point indicates the importance of obtaining high efficiencies over the whole engine operating range. From this standpoint, there is appreciable research and development to be done, particularly in regard to the diffuser, combustor, afterburner, and nozzle. High diffuser, compressor, turbine, and nozzle efficiencies over a wide range of conditions become increasingly harder to maintain as airplane maximum flight speed is increased. The question of matching these engine parts over the whole range of thrust required should be the subject of intensive research and development. Combustor and afterburner efficiencies, as pointed out previously, tend to decrease as altitude is increased, particularly at the lower airplane speeds.

Effect of Airplane Speed and Turbine-Inlet Temperature on Thermal Efficiency

The effect of engine over-all pressure ratio on thermal efficiency can be expressed as:

$$\eta_{th} = f\left[1 - \left(\frac{1}{P_{r,e}}\right)^{\frac{\gamma-1}{\gamma}}\right] \tag{17}$$

in which over-all pressure ratio, $P_{r,e}$, is the product of ram pressure ratio and compressor pressure ratio at the flight condition under consideration. The effect of airplane speed on ram pressure, assuming full recovery, is given in [redacted] VI.

[redacted]

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TABLE VI. - EFFECT OF AIRPLANE SPEED ON RAM PRESSURE RECOVERY

Airplane speed, M_a	Full-ram-recovery pressure ratio
0.5	1.2
1.0	1.9
1.5	3.7
2.0	7.8
2.5	17.1
3.0	36.3

Since both ram pressure ratio and compressor pressure ratio (fig. 22) are functions of airplane speed, thermal efficiency is also a function of airplane speed, as shown in figure 26. Curves are shown for several values of turbine-inlet temperature T_4 of a nonafterburning engine and for one turbine-inlet temperature (1540° F) of an afterburning engine. For the nonafterburning engine, two sea-level static compressor pressure ratios, 6 and 12, are used. With the afterburner in use, the effect of compressor pressure ratio over the range shown is negligible. The curves show essentially three points: (1) For the nonafterburner engines, thermal efficiency is increased by going to higher turbine-inlet temperatures. (2) For the nonafterburner engines, as airplane speed is increased thermal efficiency passes through a maximum. The speed at which this maximum occurs is increased as either turbine-inlet temperature is increased or compressor pressure ratio is decreased. (3) Thermal efficiency of the afterburner engine is lower than that of the nonafterburner engine, except at airplane speeds materially in excess of that at which the previously mentioned maximum occurs. This maximum results from the fact that as airplane speed is increased, over-all engine compression ratio increases (at $M_a = 2.0$ and an altitude of 35,000 feet to about 30 and 60, respectively, for compressors of 6 and 12 sea-level static compressor ratio) with a resultant increase in compressor-outlet temperatures. For a given turbine-inlet temperature, this successively higher compressor-outlet temperature permits successively smaller amounts of fuel to be burned, with consequent reduction in energy input. Since the quantitative losses in the compressor and the turbine do not decrease, a point is reached at which the increasing ratio of these losses to energy input become dominant in regard to thermal efficiency.

In figure 27, the data are replotted using for each flight condition the sea-level static pressure ratio that gives optimum thermal efficiency. Up to $M_a = 2.0$, the curves are essentially those of figure 27 for the pressure ratio of 12. From $M_a = 2.0$ to $M_a = 3.0$, the sea-level static pressure ratio is decreased from about 12 to a little less than 6. In this figure, the relative effects of change in turbine-inlet temperature and use of an afterburner are clearly brought out.

Propulsive Efficiency

Propulsive efficiency of the engine is the ratio of work done per unit time on the airplane (that is, engine thrust or airplane drag times the distance flown per unit time) divided by the kinetic energy produced in the exhaust gas while the airplane is traveling this distance. Using as the distance the distance traveled in unit time, the propulsive efficiency η_p is given by:

$$\eta_p = \frac{Fv_a}{1/2\ w_{ar}v_j^2 - 1/2\ w_{ar}v_a^2} \tag{18}$$

In equation (18), the effect of fuel mass is neglected.

Since

$$F = w(v_j - v_a) \tag{19}$$

$$\eta_p = \frac{2v_a}{v_j + v_2} \tag{20}$$

Figure 28 shows the effect of airplane speed on engine propulsive efficiency. The curve shows that use of the afterburner affects propulsive efficiency, but not nearly as much as airplane speed does. There is also a relatively smaller effect due to increasing turbine-inlet temperature.

Over-All Efficiency

Over-all efficiency of the engine η_e is the product of thermal and propulsive efficiencies. Figure 29 shows the effect of flight speed on over-all efficiency.

The values of F/w_{ar} listed at $M_a = 1.8$ are the thrust outputs of the engines represented in terms of thrust per pound of air flow through the engine. If the engines under consideration all have the same compressor tip diameters, the specific engine weights at the airplane speed of $M_a = 1.8$ are inversely proportional to these values of F/w_{ar} if account is taken of whether or not the engine has an afterburner. An afterburner is considered to add 25 percent to engine weight. The specific weights so computed, relative to the specific weight of an engine operating at current combustion temperatures, are listed in table VII.

It is noted that up to an airplane speed of $M_a = 2.5$ increasing the turbine-inlet temperature decreases the overall efficiency. This effect results from a decreasing propulsive efficiency with increasing turbine-inlet temperature.

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TABLE VII. - RELATIVE SPECIFIC ENGINE WEIGHTS AT $M_a = 1.8$

FOR DIFFERENT TURBINE-INLET TEMPERATURES

AB.	T_4, °F	T_5, °F	Relative specific weight
Yes	1540	3040	1.00
Yes	2040	3040	.90
Yes	2540	3040	.85
No	1540	----	2.00
No	2040	----	1.26
No	2540	----	.96

Additional data indicative of the effect of turbine-inlet temperature on specific engine weight are shown in figure 30. Values of thrust per pound of air are given as a function of turbine-inlet temperature and airplane speed. In determining figures 29 and 30, a variable inlet and a convergent-divergent nozzle (figures 23 and 24) were assumed and the appropriate efficiency and thrust losses included.

In the $M_a = 1.5$ to $M_a = 2.0$ range, if a turbine-inlet temperature of 2500° F can be used, specific weight of a nonafterburner engine will approximate the specific weight of the afterburner engine using current temperatures. For this higher temperature, the nonafterburner engine will have an over-all efficiency 50 percent higher but, as will be shown later, specific area will be increased 20 percent.

The previous discussion has indicated that turbine-inlet temperature of 2000° F or more in engines of contemporary design cannot be assured through the use of new turbine blade materials. Turbine cooling at present offers more chance of permitting such temperatures than improved materials. In the calculations for table VII and figure 30, no allowance is made for engine performance losses accompanying the cooling that would be required to permit these temperatures with current turbine materials.

Cooling as a Means of Permitting Higher Turbine-Inlet Temperatures

The operation of turbojet engines at the higher turbine-inlet temperatures will require cooling of some engine components beside the turbine so that the material temperatures will not exceed that permissible for the stresses imposed. These additional parts are the combustion chambers, the tailpipe section (including the inner "bullet"), the afterburner shell, and the exhaust nozzle which may include a plug for varying area and divergence. The air used for cooling (either air bled from the compressor or ram air) is diverted from the main thermodynamic cycle. Heat is transferred to this air during the cooling process and the cooling air is then mixed with the other air in the jet nozzle and expelled. In some respects, the cycle of a cooled engine is similar to that of a bypass engine. Any loss in engine efficiency caused by air-cooling results from pressure losses in the cooling air.

For the non afterburner engine at any turbine-inlet temperature, temperature of the gases at the exhaust nozzle will be lower for the air-cooled turbojet engine than for the uncooled engine because of the dilution of the combustion gases by the cooling air. Because of this lower temperature, thrust level of the cooled engine will be lower than that of the uncooled turbojet engine (about 1 percent for each 1 percent of compressor bleed air).

Much experimental work has been and is being conducted, notably at the Lewis laboratory of the NACA, on means of cooling turbine nozzles and blades without using excessive amounts of cooling fluid. The use of liquid coolant is being worked on but not extensively. Some of the results of the research on air cooling, using the blade type shown in figure 21, are shown in figure 31. The results in figure 31 are applicable to airplane speeds up to $M_a = 2.5$. At higher speeds, either more cooling air is required or there must be intercooling of the cooling air between the compressor outlet and the turbine inlet. It is seen that under the conditions tested, cooling-air quantities of about 10 percent of the total engine air are required to permit turbine-inlet temperatures of 2500° F and at the same time maintain the turbine blades at current blade temperatures.

Sufficient knowledge is probably now available to permit design of air-cooled turbines and turbine blades suitable for turbine-inlet temperatures of 2000° F. Additional research and development will probably provide means of achieving turbine-inlet temperatures of 2500° F at airplane speeds up to $M_a = 2.5$.

Besides offering the improved cycle efficiencies associated with higher turbine-inlet temperatures, cooling also provides the means of permitting increased gas flow capacity through the turbine, as discussed previously in connection with figure 19, and of increasing the permissible turbine stresses as discussed in relation to figures 20 to 22.

Estimates of losses in thrust and efficiency incurred by the necessary cooling-air expenditures considering the engine as a whole are shown in figure 32. The data are for an engine with a sea-level pressure ratio of 12 to 1, a turbine-inlet temperature of 2040° F, and an afterburner temperature of 3040° F. Cooling air for the afterburner shell is assumed to be supplied at ram pressure, and compressor discharge bleed air is diverted to cool the turbine and the exhaust nozzle plug. The data of figure 32 are based on an engine designed to current values of turbine centrifugal stress. [redacted]

The solid curves represent the calculated performance of the engine based on the temperatures specified, but with no diversion of cooling air to any of the components; in other words, the high temperature performance was assumed to be obtained without any penalty for cooling. The dotted curves represent the performance with cooling penalties assessed in accordance with the estimated cooling loads that would be imposed on the cycle. The loss in thrust per pound of compressor air flow is seen to be less than 5 percent in all cases; the loss in over-all engine efficiency is inconsequential at subsonic velocities, and increases to 4 per-
3460 cent at $M_a = 2.0$ and 9 percent at $M_a = 3.0$. These penalties are not large in comparison to the improvements in performance that are obtained with high cycle temperatures at the higher Mach numbers.

For higher turbine-inlet temperatures (2500° F), the cooling requirements become more severe, and practically all components exposed to the combustion gases will require cooling. Experimental investigations conducted at the Lewis laboratory with various combustor designs indicate that major changes in design will not be required to permit operation at turbine-inlet temperatures up to 2500° F. No experimental information on cooling has been obtained at these higher temperatures. Simple convection cooling designs with air ducted through an annular section can be used for tailpipe cooling of nonafterburning engines. Ram air can also be used as the cooling medium for most applications. If high temperatures at low flight speeds are desired, compressor bleed or exhaust ejectors will be required. The air quantities required are small, and the heated cooling air can be expended through a nozzle to obtain thrust.

In considering higher turbine-inlet temperatures, attention must be given to the effect of this type of engine operation on cruise (part-throttle) operation. For an analysis of this effect, an interceptor is considered that is designed for combat at 50,000 feet and $M_a = 2.0$ and for cruise at 35,000 feet and $M_a = 0.9$. It is further assumed that the L/D at combat is one-fourth that at cruise; that is, the required combat thrust is four times the cruise thrust. These values are representative of current interceptor design (see fig. 10).

Two engines are considered: one operating under current conditions, that is 1540° F turbine-inlet temperature and 3040° F afterburner temperature; the second, a nonafterburner engine with a turbine-inlet temperature of 2540° F. At the cruise altitude and Mach number and at various engine throttle settings, the ratio of available cruise thrust to full throttle thrust at the combat flight condition is computed. The results are shown in figure 33. In these data, the turbine cooling losses have been included. With the afterburner engine, afterburner control from zero augmentation to full augmentation is assumed. The data show cruise efficiency of the high turbine-inlet temperature nonafterburner engine to be about one-tenth less than that of the afterburner engine. The high engine efficiency of the nonafterburner engine at full throttle is again emphasized.

[redacted]

Additional estimates on the effects of higher turbine-inlet temperatures are presented in table VIII. The data apply to the two engines considered in figure 33 and an afterburner engine with a turbine-inlet temperature of 2540° F and an afterburner temperature of 3040° F. Estimated values of engine efficiency, specific weight and relative frontal area at two flight conditions are presented. The table again shows that high turbine-inlet temperature decreases engine frontal area and specific weight of the afterburner engines. It also shows the increase in frontal area for a given thrust that accompanies use of high turbine-inlet temperatures without an afterburner. 3460

Because of the apparent improvements in engine performance obtainable through use of turbine cooling, additional detailed research and development is required on problems associated with other engine components that are posed by its use. Additional development effort is also needed on the problems presented in using turbine cooling to assist in achieving the three objectives discussed in relation to it: (1) greater gas flow rates through the turbine, (2) higher turbine rotational speeds, and (3) higher turbine-inlet temperatures.

TABLE VIII. - EFFECT OF AFTERBURNER AND TURBINE-INLET

TEMPERATURE ON PERFORMANCE

Flight condition	SLS Pressure ratio, 12				SLS Pressure ratio, 6		
	Engine	(1) AB.	(2) AB.	(3) No AB.	(1) AB.	(2) AB.	(3) No AB.
M_a 2.0 50,000 ft Combat thrust	η_e, percent Rel. sp. eng. wt Rel. area	24 1.00 1.00	30 0.84 .84	42 1.04 1.22	24 0.98 .98	28 0.91 .92	36 1.04 1.22
M_a 0.9 35,000 ft $\frac{1}{4}$ Combat thrust	η_e, percent	23*	20*	21	18*	16*	17

Engine temperatures at combat, °F	Engine	(1) AB.	(2) AB.	(3) No AB.
	Turbine inlet	1540	2540	2540
	Afterburner	3040	3040	----

*Afterburner off.

[redacted]

Fuel Heat of Combustion and Maximum Combustion Temperatures

An increase in fuel heat of combustion gives a directly proportional increase in range (eqs. (3) or (4)). Increasing the combustion temperature increases thrust per pound of air and therefore decreases specific weight of the engine; or, as generally considered, increases thrust output of a given engine.

Heat of Combustion

The determination of those fuels which will produce chemical heats of combustion higher than those obtained with conventional hydrocarbon jet fuels can best be made by plotting the atomic number of the elements against the heats of combustion of the elements, figure 34. The expected periodic variation in such a plot of a chemical property is obtained. For reference, the approximate heat of combustion for the current hydrocarbon fuels (JP-4), 18,500 Btu per pound, is included. The data show that the elements of interest with respect to high heat of combustion, in addition to hydrogen and carbon, are lithium, beryllium, and boron. The heat of combustion of lithium is not sufficiently higher than that of carbon to make lithium of much interest. Beryllium will not be considered because of its scarcity and extreme toxicity. In addition to hydrogen and the hydrocarbons, then, boron and the hydrides of boron are the fuels on which emphasis should be placed. Two boron hydrides, diborane (B_2H_6), and pentaborane (B_5H_9), have been under investigation for some time. Under normal atmospheric conditions diborane is a gas and therefore poses the inherent difficulties associated with the use of a gaseous fuel. Pentaborane, a liquid of about 15 percent lower density than the liquid hydrocarbon fuels, has, as indicated, a heat of combustion about 1.5 times that of hydrocarbons. Pentaborane, therefore, offers a potential range increase of about 50 percent. The disadvantages of the fuel are that it is dangerously toxic, it is unstable, it has solid combustion products, and it is expensive to manufacture.

Under the auspices of the Bureau of Aeronautics, United States Navy, an investigation is under way to determine to what extent a fuel of satisfactory properties, but with a heat of combustion approaching that of pentaborane can be synthesized from the elements hydrogen, boron, and carbon. The ternary chart in figure 35 shows the effect of the relative proportions of these three elements on the heat of combustion of such a fuel. The calculated lines of constant heat of combustion shown in the chart, together with the measured values for diborane and pentaborane, were obtained from reference 9; the value for acetylene was obtained from the National Bureau of Standards; other measured values were supplied by the Mathieson Chemical Corporation. The calculated lines show the general trend of varying the percentage composition on heat of combustion, but the values given by the lines do not necessarily apply to each of the individual compounds listed; for example, acetylene.

Research has shown that adding hydrocarbon constituents to the boron-hydrides does much to reduce cost and improve handling qualities of the fuel by decreasing toxicity. The chart (fig. 35) shows that certain such fuels, decaborane-ethylene for instance, have heats of combustion about 1.4 times that of current jet fuel. In addition, these components have better physical properties and less toxicity than does pentaborane. 3460

When a boron fuel is burned in the primary combustor of an unmodified turbojet engine, solid combustion products (boron oxides) are formed ahead of the turbine; these products tend to deposit to a prohibitive extent in the combustion chamber and on the turbine nozzles and blades. Elimination of this problem will not be easy, but progress is being made.

Use of a borane fuel in an afterburner with a conventional hydrocarbon fuel in the primary combustors would, at $M_a = 2.0$, give about a 25 to 30 percent effective heat of combustion increase over that using JP-4 in the afterburner and therefore range increase 25 to 30 percent. Another advantage of the boron-hydride fuel is that its combustion efficiency is depreciated less by increase in altitude than is that of hydrocarbon fuel. The extent to which combustion efficiency of hydrogen-boron-carbon fuel will be affected by altitude is not known.

Maximum Combustion Temperatures

The elements that should be considered as fuels that burn to higher temperatures and thus produce additional thrust augmentation are indicated in figure 34. Aluminum or magnesium give appreciably higher combustion temperatures than do the hydrocarbons. Silicon and phosphorus should have about the same values as aluminum and magnesium. Boron shows some improvement over the hydrocarbons. It is questionable that other elements are of interest, because of the generally lower heats of combustion at atomic numbers above that of phosphorus.

Both magnesium and aluminum have appreciably lower heats of combustion than do the hydrocarbons although their combustion temperatures are higher. This fact means that their high combustion temperatures are obtained at the expense of high fuel-air ratios and therefore high specific fuel consumptions (that is, lower values of $h\eta_e$ even though η_e remains high).

[redacted]

The relation between combustion temperature and thrust is given by the equations:

$$F = w_{ar} \left[(1 + f/a) \; v_j - v_a \right] \tag{21}$$

in which

F thrust

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w_{ar} weight flow of air per unit time

f/a fuel-air ratio

v_a airplane velocity

v_j jet velocity

and

$$v_j = K \sqrt{\frac{T}{m} \left[1 - (P_{r,x})^{\frac{\gamma-1}{\gamma}} \right]} \tag{22}$$

in which

K a constant

T combustion temperature

m average molecular weight of exhaust products

$P_{r,x}$ expansion ratio of gases

Augmentation values computed for the fuels of interest are shown in table IX. The special fuels are assumed to be used in the afterburner only, with JP-4 being used in the combustor. The data show that substitution of other fuels for JP-4 in the afterburner of a turbojet engine may produce theoretical increases in augmentation of up to 40 percent. Research on powdered magnesium slurries in JP-4 is being conducted with considerable success. Over-all fuel consumption rate would be about the same for a 40-percent augmentation ratio, whether the augmentation is obtained by use of magnesium as a supplementary fuel or by use of auxiliary rockets.

[redacted]

[redacted]

TABLE IX. - EFFECT OF AFTERBURNER FUEL ON RELATIVE

TURBOJET AUGMENTATION, AIRPLANE SPEED

$M_a = 0$ TO $M_a = 1.0$

Fuel	Thrust ratio, $\frac{\text{Aug thrust}}{\text{Aug thrust}_{C_8H_{16}}}$	Afterburner combustion, T_5, °F**	sfc, lb fuel-hr/ lb thrust
C_8H_{16}*	1.00	3700	2.1
H_2	1.04	3700	1.2
B	1.11	4600	2.7
B_2H_5	1.09	4300	2.1
Al	1.27	5200	5.3
Mg	1.42	6000	6.3

*Representative of JP-4.

**Assuming stoichiometric mixture.

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FUEL AVAILABILITY

The availability of turbojet fuel is determined by the amount of the crude products that are available, the percentage of the products that can be allotted to production of turbojet fuel, the cost in men and materials of the refining or manufacturing process, and the limits on quantity of the finished products imposed by the fuel specification.

In regard to the special fuels mentioned, the availability of boron is probably sufficient for specialized use. Based on present knowledge, the cost of manufacturing a hydrogen-boron-carbon fuel is high and may be the controlling factor in determining the extent of its use. Magnesium is sufficiently plentiful for magnesium slurry in petroleum fuel. Cost of the powder in quantity production will probably be about $0.50 a pound.

Figure 36 shows the availability of petroleum fuel, currently JP-3 and JP-4, in relation to the respective fuel specifications. The figures are based on the finished product in relation to the total of crude oil available and do not consider the many other uses for which the

[redacted]

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components of JP-3 and JP-4 might be required in event of an all-out emergency. Total crude oil available, in case of an all-out national emergency, is estimated to be 8 million barrels a day. The extent to which the amount of turbojet engine fuel available may be decreased by more restrictive specifications is indicated in the comparison of the figures for JP-4 with those for JP-3. Current thinking is that such a decrease in fuel availability is advisable in order to increase fuel quality with consequent lessening of engine maintenance. Successively higher flight speeds at high altitudes, with the accompanying higher temperatures to which fuel in the airplane will be subjected, will require further elimination of the lower boiling temperature (gasoline) constituents and will thus further decrease quantity available.

Because of the other fuel requirements, all the fuel indicated in the turbojet fuel barrel will not be available. Estimates are that 25 percent of the total crude, that is, 2 million barrels per day of jet fuel, will be allowable. Assuming an average engine efficiency of 20 percent, 2.0×10^6 barrels of JP-4 a day would provide 3×10^{11} pound thrust-miles a day. Assuming, for example, an average airplane weight of 50,000 pounds and an average airplane L/D of 10, this fuel quantity represents 6×10^7 airplane-miles a day.

LUBRICATION

Provision of adequate lubrication for the turbojet engine at high flight speeds presents problems that will be solved only through intensive research and development. At Mach 2.0, the stagnation temperature (NACA standard day) is 240° F; at Mach 2.5, 400° F; and at Mach 3.0, 600° F. The magnitude of the lubrication temperature problem will depend upon the duration of high-speed flight time. If this time is long enough that the temperature of the aircraft approaches the stagnation temperature, the problem is severe.

It becomes apparent that unless mechanical refrigeration is used lubricants and bearings must be developed to withstand higher temperatures. Table X shows bulk lubricant and bearing temperatures based on assumptions that bulk lubricant temperature is 50° F higher than stagnation temperature and bearing temperature is 150° F higher.

TABLE X. - ESTIMATED EFFECT OF AIRCRAFT SPEED ON BULK

LUBRICANT AND BEARING TEMPERATURE

Mach no.[a] M_a	Bulk lubricant temperature, °F	Maximum bearing temperature, °F
2.0	290	390
2.5	450	550
3.0	650	750

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[a]Sustained flight above 35,000 feet.

The presently used bearing material, SAE 52100 steel, is limited to highly loaded bearing applications below 350° F. Above this temperature, two changes take place that make it unsuitable. Structural changes take place that cause permanent dimensional increases, and the material softens. Proper heat treatment can raise the dimensional stability limit of SAE 52100 to 400° F; the material will, however, be softer than is desirable. Tool steels have sufficient hardness and dimensional stability up to 800° F; to date, however, there are insufficient data available to tell if they will have satisfactory fatigue life.

Friction between steel and the material currently used for bearing cages, iron-silicon-bronze, is almost constant up to 600° F. As temperature is increased above that value, friction increases considerably. Use of other materials, such as nodular iron or certain of the nickel alloys, whose coefficient of friction decreases with increase in temperature, will possibly alleviate this problem. Past difficulties with cages at moderate temperatures indicate that development of adequate cages for high temperatures will require substantial effort.

Provision of a lubricant that will operate satisfactorily at the higher temperatures also appears difficult. Present synthetic fluids, such as the diesters, are adequate for bearing temperatures up to 500° F, if bulk lubricant temperature does not exceed 350° F. If a closed system is used that limits the amount of oxygen that may contact the lubricant, allowable bulk lubricant temperature may be increased to 500° F. For bearing temperatures above 500° F, it seems probable that solid or gaseous lubricants will be required.

CRITICAL MATERIALS

The section on engine efficiencies discussed the development of engine materials to withstand higher turbine-inlet temperatures. This

[redacted]

section discusses the availability of materials in relation to turbojet engine construction. The materials whose availability may limit the aircraft-engine production are the elements cobalt, columbium, chromium, nickel, molybdenum, and tungsten. For current engine usage, titanium is not limiting. This situation can change markedly with decreased titanium production costs and consequent increased demands.

Figure 37 shows the way in which each of these limiting or critical materials is distributed among the major components of turbojet engines. The values are averaged from data for the engines produced in the United States. Cobalt, columbium, molybdenum, and tungsten are used only in the turbine and afterburner, but chromium and nickel are distributed fairly uniformly among all the components. The total weight of these critical materials is generally about 17 percent of the gross engine weight for either nonafterburner or afterburner engines.

In unpublished data, W. H. Woodward of the NACA staff has shown (fig. 38) that for the J-33, J-34, J-35, J-40, J-42, J-47, and J-48 engines (represented by "diamonds") the distribution of the critical materials can be conveniently shown in graphical form. In each case, the amount of each critical material, or group of materials, is given as a percent of the total weight in the engine of all the materials listed in the figure. For these earlier turbojet engines, columbium and cobalt were the metals that would limit engine production in case of a national emergency (ref. 10). The data represented by the three squares are for more recent engines, in which the manufacturers have used but little cobalt and columbium. To the extent that these three points represent current design procedures, they show that cobalt and columbium can be reduced to 2 percent or less of the total critical material. They also indicate that as the ratio of nickel to chromium is increased, the cobalt-columbium-tantalum percentage and the molybdenum-tungsten percentage decrease.

The + symbol in figure 38 indicates the target for the average distribution of the materials, considering all engines produced, set in 1952 by the then existing Munitions Board (ref. 10). This analysis allowed for use of these critical materials up to 10 percent of the total engine weight; compared to the current value of 17 percent.

Nickel and chromium are currently believed to be the materials that would limit production of aircraft turbojet engines in an all-out national emergency. Chances are slight that the amounts of these materials that can be mined within the United States and adjacent territories will increase beyond present estimates. The data presenting the relative values of requirements and availability were assembled in 1952 (ref. 10).

In table XI, using the 1952 values, the amounts of virgin nickel and virgin chromium required for each engine and the estimated total amounts

[redacted]

[redacted]

available for turbojet engine production are tabulated. The amounts available are based on conditions of maximum permissible availability from North America and the adjacent islands and include such stockpiling as was then planned. Under current practice, the amount of virgin nickel or chromium required to build an engine is roughly four times the amount that remains in the finished engine. As shown in the second part of the table, it is estimated that with rigid scrap and discarded parts control this ratio could be decreased to about 2 to 1, with a corresponding increase in the number of engines produced. The 5000-pound engine used as an example would give a sea-level static military rated thrust of 11,000 to 15,000 pounds, approximately that of current engines. 3460

TABLE XI. - ESTIMATED ENGINE PRODUCTION AS LIMITED BY NICKEL AND CHROMIUM

	Nickel	Chromium
Virgin metal available/year (5 yr period) lbs	108×10^6	190×10^6
Under current practice		
Virgin metal required/engine, percent engine weight	30	38
Losses:		
Mill and melt, percent engine weight	6	7
Fabrication, percent engine weight	17	22
Virgin metal in finished engine, percent engine weight	7	9
No. 5000-lb engines/year	70,000	100,000
With rigid scrap and discarded parts control		
Virgin metal required/engine, percent engine weight	15	19
Losses:		
Mill and melt, percent engine weight	5	6
Fabrication, percent engine weight	3	4
Virgin metal in finished engine, percent engine weight	7	9
No. 5000-lb engines/year	140,000	200,000

[redacted]

The practicability of considering other elements as replacements for the six elements listed as critical materials can be evaluated from a consideration of a plot of melting temperature against atomic number for the various elements, together with information on the availability of the elements. These data are given in figure 39. Again, the periodic variation of the property under examination is noted. The relative abundance figures indicated by the symbol code represent current world production, and the values at the plotted points (from ref. 11) indicate
3460 the percentage of current world production of the material that is mined in North America and the adjacent islands. For a material to be considered for more than 5 percent use in the complete turbojet engine, the relative abundance figure should be 15 to 50, or greater, and the percentage mined in North America 10 percent or greater.

Metals are required that have a melting temperature equal to or above that of the nickel-cobalt-chromium group and are about as available as these metals. The only higher melting elements that are sufficiently available to warrant consideration are molybdenum and tungsten. Alloys containing these materials have and are being worked on. The strengths of tungsten alloys have so far been disappointingly low. The molybdenum alloys appear more favorable; but unless means of reducing their oxidation are found, their satisfactory use cannot be assured. There are only two additional elements that have melting temperatures in the range of those of nickel, cobalt, and chromium: vanadium and titanium. Development has been carried out on vanadium alloys, but so far the brittleness of high-temperature vanadium alloys has not been eliminated. Work on this metal is continuing. Although research on titanium is proceeding, its characteristics as a major constituent of high-temperature alloys are still unknown.

The data in figure 39 indicate that further relief in the critical material situation through substitution of materials cannot be assured.

Cermets (mixtures of ceramics and metals) and intermetallics are being investigated as materials for turbine blades. Work is being done to obtain turbine blades that will permit turbine-inlet temperatures of 2000° F or higher, but success is far from certain.

If figures presented in table XI do not show a sufficient number of pounds of aircraft engines under national emergency conditions, a reexamination of the materials situation is in order.

CONCLUSIONS

Airplane Performance

The results of this analysis have shown that for combat airplanes with weight distrib[redacted]tly in use, airplane performance is affected by the major airplane [redacted]ngine variables as follows:

[redacted]

1. Except as airplane take-off or landing runs is limited or as range is of secondary importance, gross weight of the airplane at take-off is largely determined by military load. In general, a gross weight of 12 to 14 times the military load gives a reasonable compromise between range and gross weight.

2. The range of each class of airplane (that is, fighter, bomber, etc.) is for the most part, determined by heat of combustion of the fuel, efficiency of the engine, and lift-drag ratio of the airplane. Range will be increased directly as each of these variables is increased. 3460

3. Where gross weight is limited by take-off or landing facilities, increase in airplane lift-drag ratio is the most effective means of permitting the required range, altitude, and speed to be maintained with continually increasing military load. For the bomber, increase in fuel heat of combustion or in engine efficiency, or decrease in specific airframe weight* are equally effective, although these changes are somewhat less effective than increase in airplane lift-drag ratio; decrease in specific engine weight* is least effective. With the fighter, decrease in specific airframe weight* is almost as effective as increase in lift-drag ratio; next in order of effectiveness is decrease in specific engine weight*, and least effective is increase in the heat of combustion of the fuel or in effieiency of the engine.

4. The most effective method of increasing permissible airplane altitude, assuming the 12 to 14 ratio between gross weight and military load is to decrease engine specific weight or to increase airplane lift-drag ratio. In either case, a 10-percent improvement results in a 2500 to 3000-foot increase in airplane ceiling. If the airplane weight distribution is not changed, a given increase in lift-drag ratio will at the same time result in a proportional increase in range.

5. Attainment of military airplane speeds greater than Mach 2.0 will for the most part be dependent at high altitudes on solution of the problems imposed by the high stagnation temperatures. The heating problems are probably equally divided between the airframe, the propulsion system, and the military load. At low altitudes, such speeds will be dependent on the rate at which higher engine pressure loads can be tolerated without increasing engine weight.

6. The size engines required, as defined by sea level military rated thrust, is dependent on (1) that flight condition which requires the highest value of the product of the ratio of thrust to airplane gross weight, and ratio of thrust available at sea level to thrust available under this condition, and (2) the number of engines in the airplane. As combat altitude is increased, the required sea-level static military rated thrust for a given gross weight airplane is increased. With current fighter airplane

*Assuming the dec[redacted] ratio of fuel to gross weight.

weight distributions and lift-drag ratios, for a combat altitude of about 60,000 feet at $M_a = 2.0$, the available take-off thrust will exceed the airplane gross weight at take-off. For current military loads, such fighters will require multiple engines or single engines delivering in excess of 20,000 pounds military rated sea-level static thrust. These high-altitude thrust demands will require that specific engine weight at take-off be about two-thirds current value. As research results in improvements in aerodynamic compressor performance at high airplane speeds and improve
3460 combat lift-drag ratio, engine size demands will lessen.

Propulsion System

The turbojet engine data analyzed in the report have indicated that the three major propulsion-system variables (specific engine weight, engine efficiency, and fuel heat of combustion) can be foreseeably improved as follows.

Engine Specific Weight

1. Specific engine weight will be decreased to about two-thirds the current value through improvements in the air-handling ability of engines. This decrease will apply through the whole flight range and will be accompanied by an equal decrease in specific engine area.

2. Further marked decreases in specific engine weight will be realized through improvements in mechanical design knowledge that allow less metal to be used in a given size engine. This phase of turbojet engine development is being emphasized heavily.

3. Specific engine weight and specific engine area at airplane altitudes in excess of 35,000 feet and at airplane speeds in excess of $M_a = 1.3$ can be decreased appreciably. If engine speed can be increased to bring the compressor to rated peripheral Mach number, the decrease will be 20 percent or more at $M_a = 2.0$ and 35 percent or more at $M_a = 2.5$. The increase in engine speed will require turbine cooling and strengthening of the compressor. The speed increases for the weight decreases given are 16 percent at $M_a = 2.0$ and 30 percent at $M_a = 2.5$.

4. As airplane speed is increased above $M_a = 1.3$, performance of the inlet air diffuser and of the exhaust nozzle become increasingly important in relation to specific engine weight. Variable-geometry and variable-area inlets and variable-area convergent-divergent exhaust nozzles are required. The general principles involved in the design of these devices have been the subject of much continuing research and development. Simple and reliable mechanical designs are required.

5. For short bursts of high thrust (that is, momentary decrease in specific engine weight) petroleum fuels containing magnesium slurries can be used. By using such fuel, a 40-percent increase in thrust may be obtained. This increase is accompanied with a trebling of specific fuel consumption and an increase of several hundred degrees in afterburner temperature. With respect to fuel consumption, rockets are a competitive means of supplying this augmentation.

Engine Efficiency 3460

1. Turbojet engine efficiency may be increased about a quarter-fold through improvements in component design, but the rate of improvement will be relatively slow.

2. With afterburner engines, increasing turbine-inlet temperature to 2000° or 2500° F will give about a 10-percent or somewhat greater increase in engine efficiency. This increase is accompanied by a somewhat lower decrease (depending on the quantity of cooling air required) in specific engine weight and frontal area. Turbine-inlet temperatures of 2000° F are probably feasible now, through use of turbine cooling. Temperatures of 2500° F appear possible through additional research and development.

3. If turbine-inlet temperatures of 2500° F can be achieved, specific weight of an nonafterburner engine in the $M_a = 1.5$ to $M_a = 2.0$ region will equal specific weight of an afterburner engine using current temperatures. The nonafterburner engine will have an efficiency 50 percent greater, but its specific frontal area will be 20 percent greater.

Fuel Heat of Combustion

1. Heat of combustion of the fuel can be increased up to 50 percent through use of boron-hydride fuels. The major problems to be overcome in the production and use of such fuels are (a) they are generally extremely toxic and are otherwise dangerous to handle, (b) they are expensive to manufacture, and (c) boric oxides in the combustion products may form solid deposits which tend to adhere to the engine parts. All three problems are being worked on extensively. It appears that a fuel consisting of hydroborons in chemical combination with hydrocarbons will greatly lessen the toxicity and general handling problem and provide a 40-percent heat of combustion increase over current hydrocarbon fuel. The cost of these fuels may determine the extent of their use.

2. Current petroleum fuel resources will allow for aircraft operation to a value of about 3×10^{11} pound thrust miles a day, under national emergency conditions. This number includes turboprops and ram-jets as well as turbojets.

Materials

Progress in regard to the materials of which the turbojet engine is made is as follows:

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Columbium and cobalt have been deleted from the engine to the extent that production of these two elements no longer limits production of aircraft engines.

In case of an all-out emergency, nickel and chromium will probably limit the rate of turbojet engine production. Current estimates are that in case of an all out emergency present methods could produce 70,000 to 100,000 engines per year over a 5-year period. There is little reason to expect that use of new materials will permit this figure to be increased. More adequate scrap control, however, might double this limit.

Current progress on the development of turbine-blade materials that will stand increases of more than 200° F in the turbine-inlet temperature is slow, and successful solutions cannot be assured. However, an increase of as much as 100° F in permissible material temperature will have important affects in increasing engine reliability, or in permitting higher engine stresses.

Lubrication

Provision of adequate lubrication for turbojet engines at stagnation temperatures occurring at speeds in excess of $M_a = 2.0$ is one of the major difficulties to be overcome if such flights are to be feasible for other than short bursts. Methods of providing this lubrication are not now apparent.

APPENDIX - SYMBOLS

The following symbols are used in this report:

A	area
AB.	afterburner
C	constant
D	drag
d	diameter
F	thrust
f	function
f/a	fuel-air ratio
g	acceleration of gravity
h	heat of combustion
J	mechanical equivalent of heat
K	constant
L	lift
M	Mach number
m	average molecular weight of exhaust products
N	compressor revolutions per unit time
P	total pressure
P_r	pressure ratio
p	static pressure
R	range
R_{ar}	gas constant for air
SLS	sea-level static
sfc	specific fuel consumption
T	total (stagnation) temperature
t	static temperature

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v velocity

W weight

w mass-flow rate

Δp pressure differential

γ ratio of specific heats

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η efficiency

ρ density

Subscripts:

a airplane

abs absolute

af airframe

alt altitude

ar air

av available at the flight condition

ax axial

C cruise

c compressor

e engine, installed power plant

eng engine as supplied by manufacturer

f fuel

g gross

j jet

m military

max maximum permissible

n annular

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o	initial flight conditions
p	propulsive
per	peripheral
r	ratio
sl	sea level
TO	take-off
th	thermal
x	expansion
0	free-stream
1	diffuser inlet
2	compressor inlet
3	compressor outlet
4	turbine inlet
5	afterburner
6	nozzle exit

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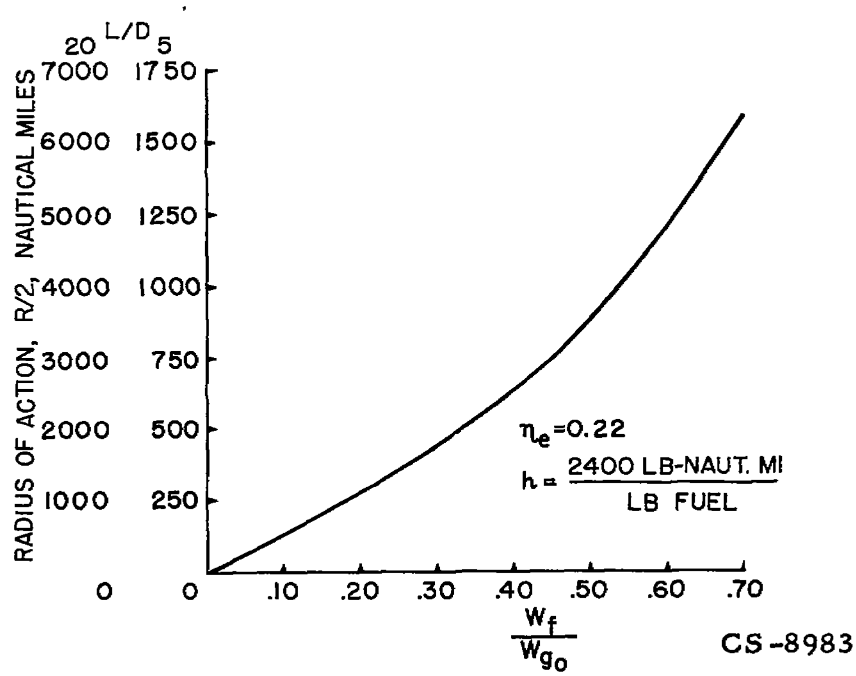

Figure 1. - Effect of cruise fuel quantity on radius of action.

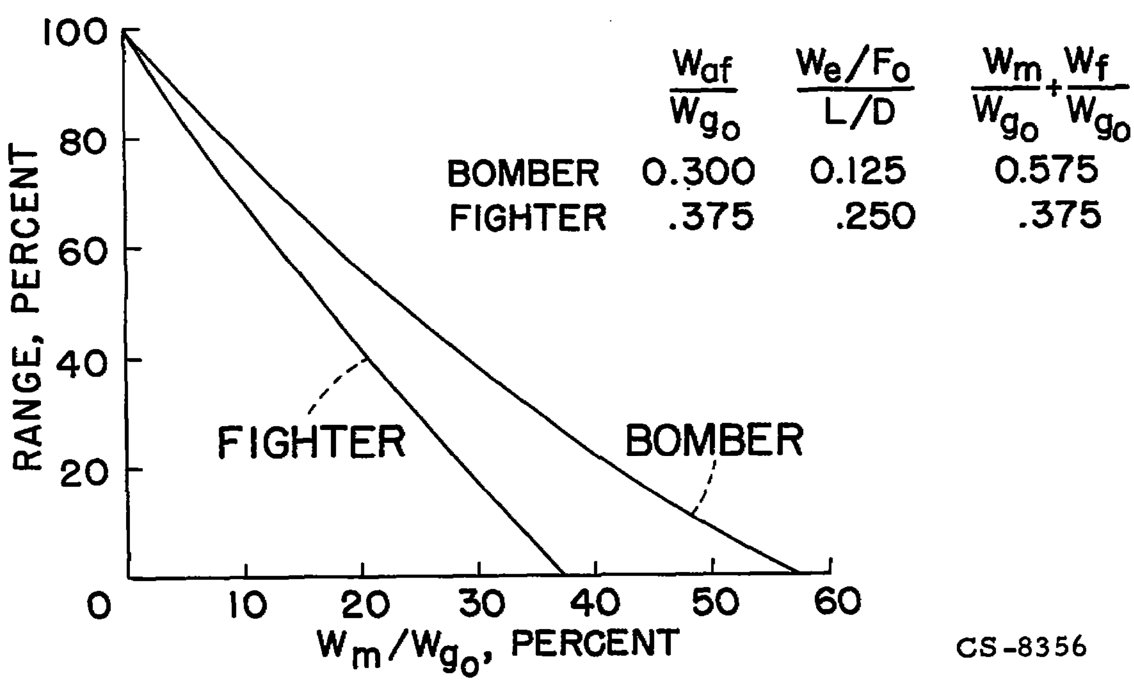

Figure 2. - Effect of relative military load on range.

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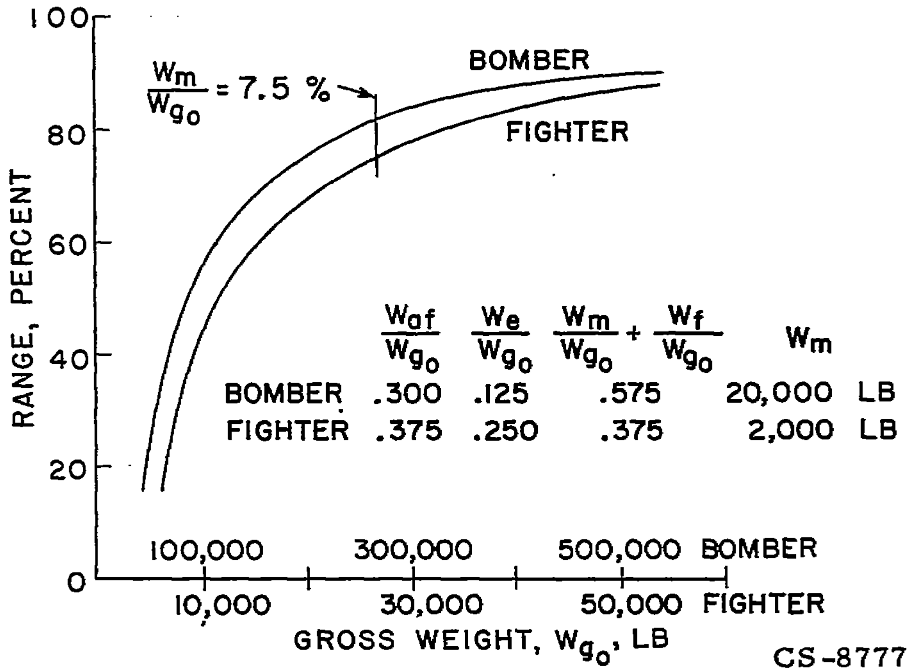

Figure 3. - Effect of gross weight on range for constant military load.

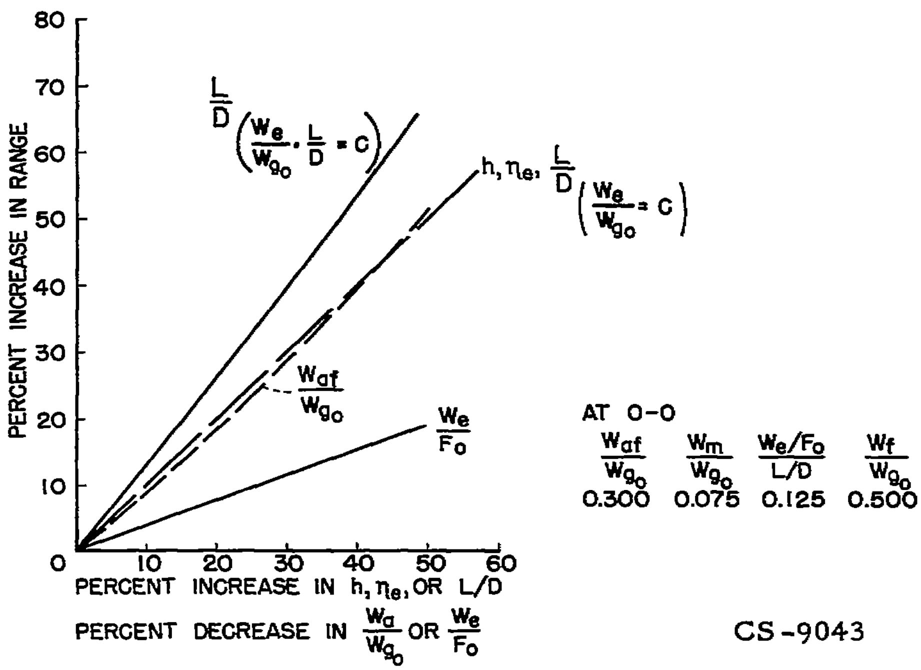

Figure 4. - Effects of airplane variables on bomber range.

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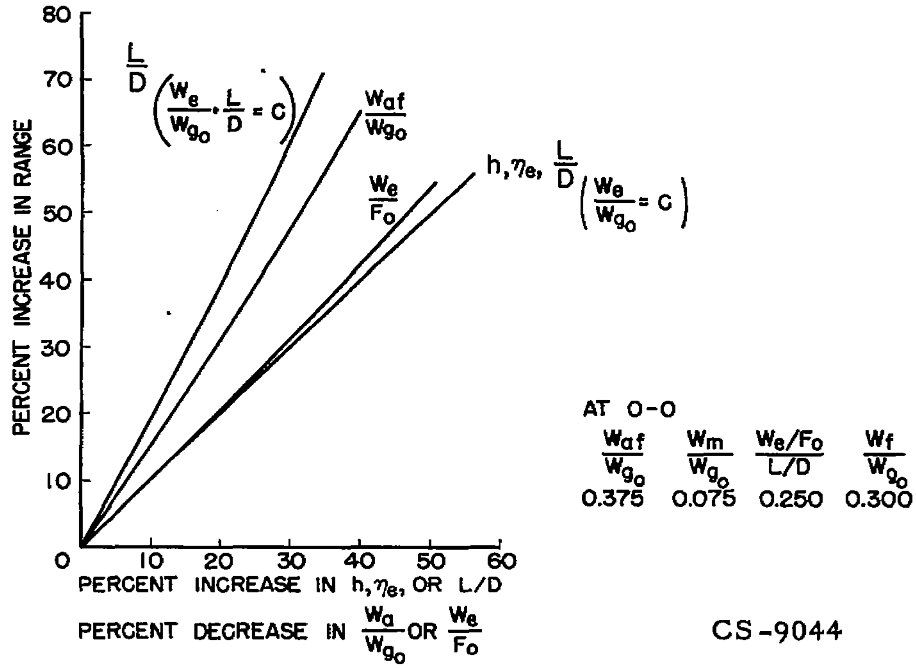

Figure 5. - Effect of airplane variables on fighter range.

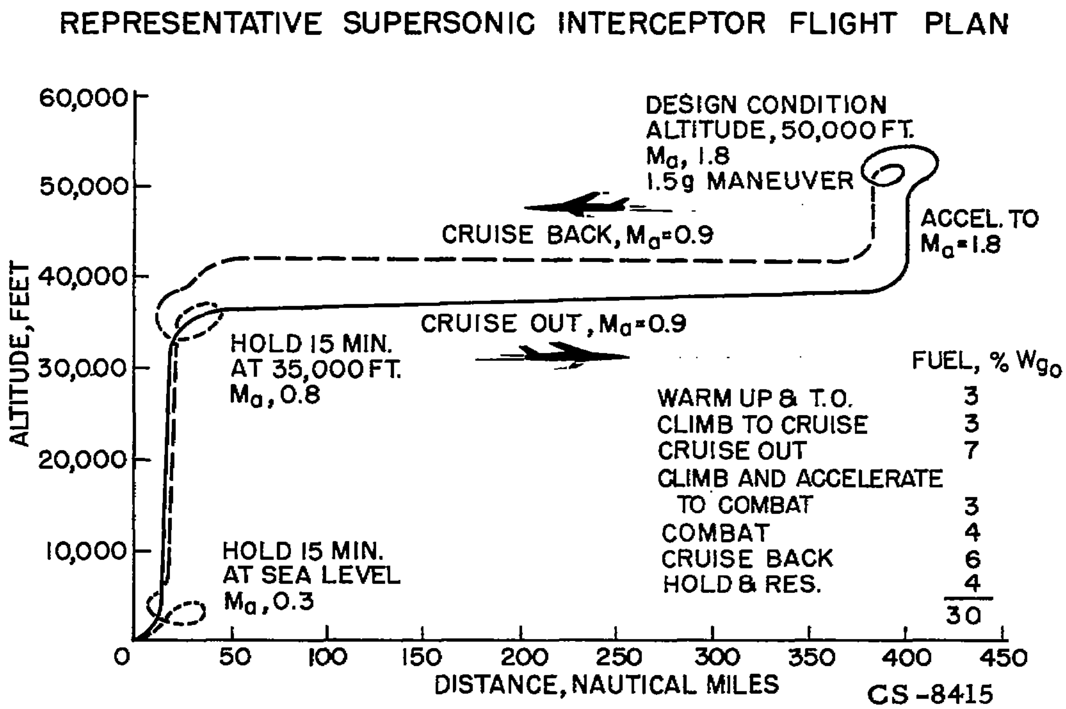

Figure 6. - Representative supersonic interceptor flight plan.

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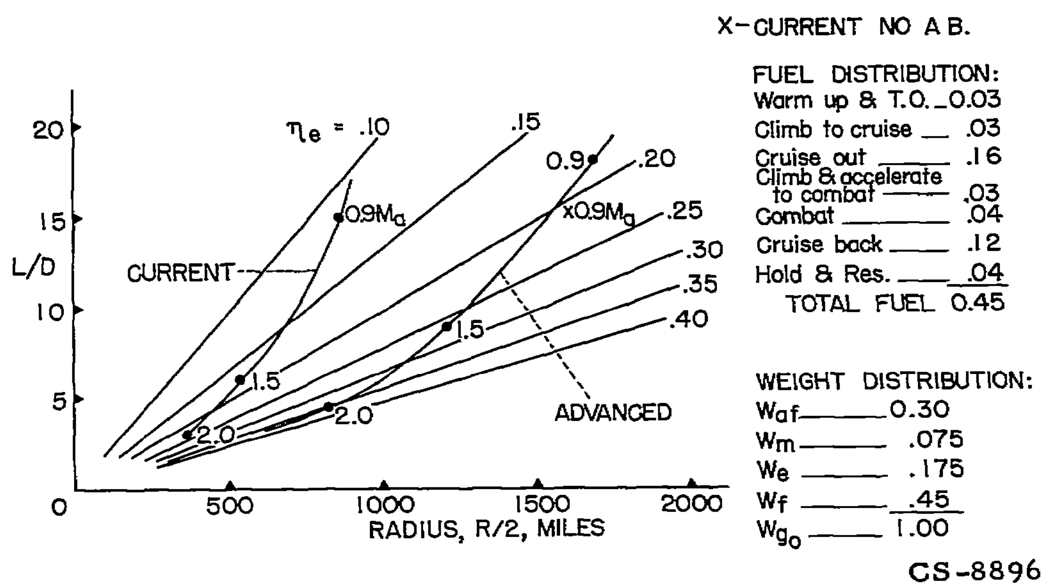

Figure 7. - Some effects of airplane L/D and engine efficiency η_e on range.

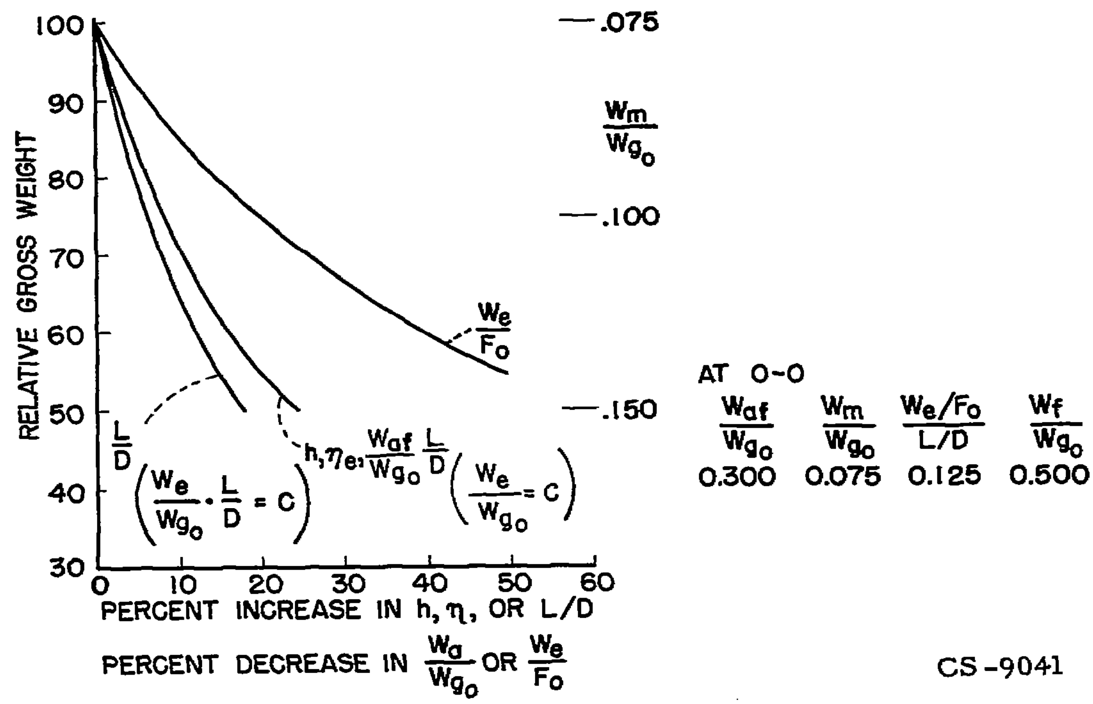

Figure 8. - Effect of airplane variables on bomber gross weight for constant range.

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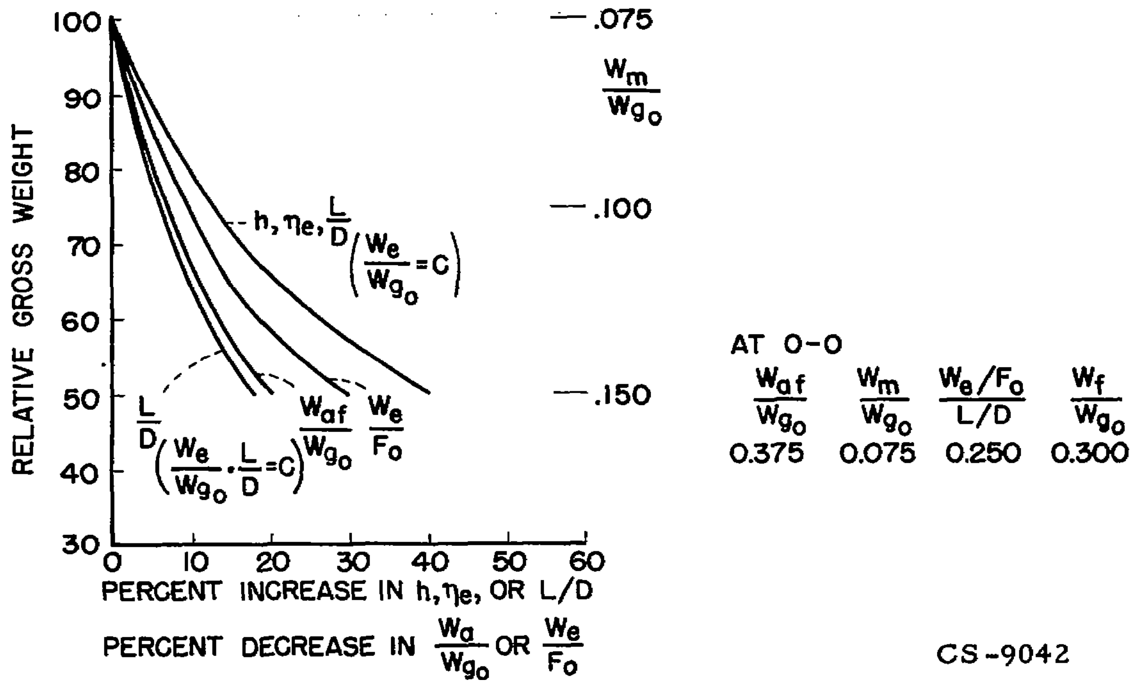

Figure 9. - Effect of airplane variables on fighter gross weight for constant range.

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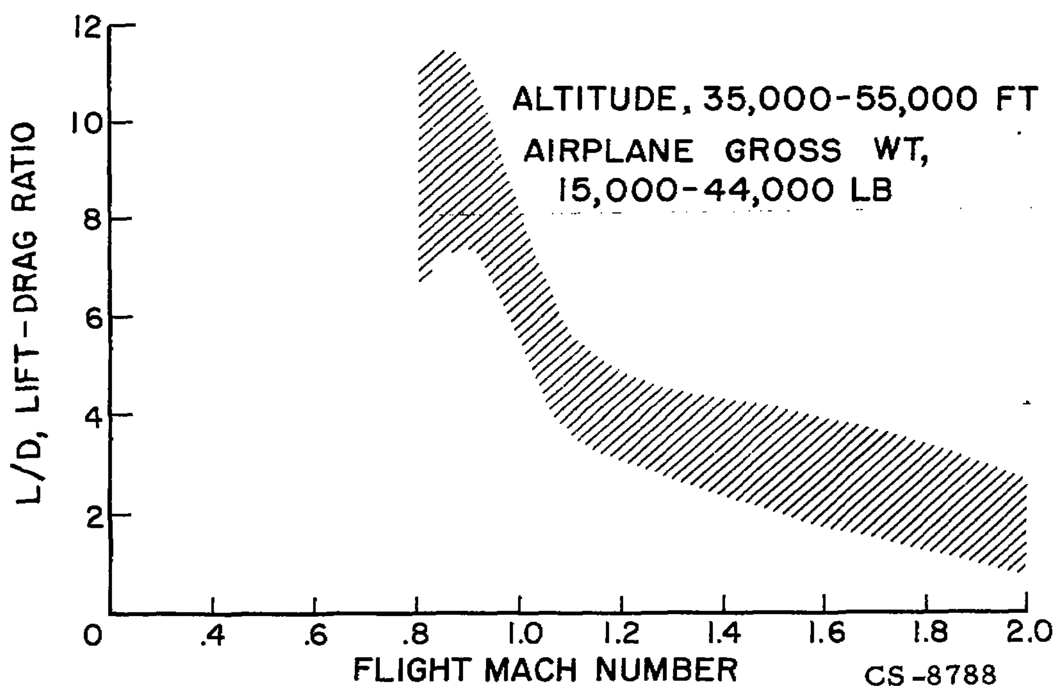

Figure 10. - Relation between fighter speed and lift-drag ratio of trimmed airplane in level flight.

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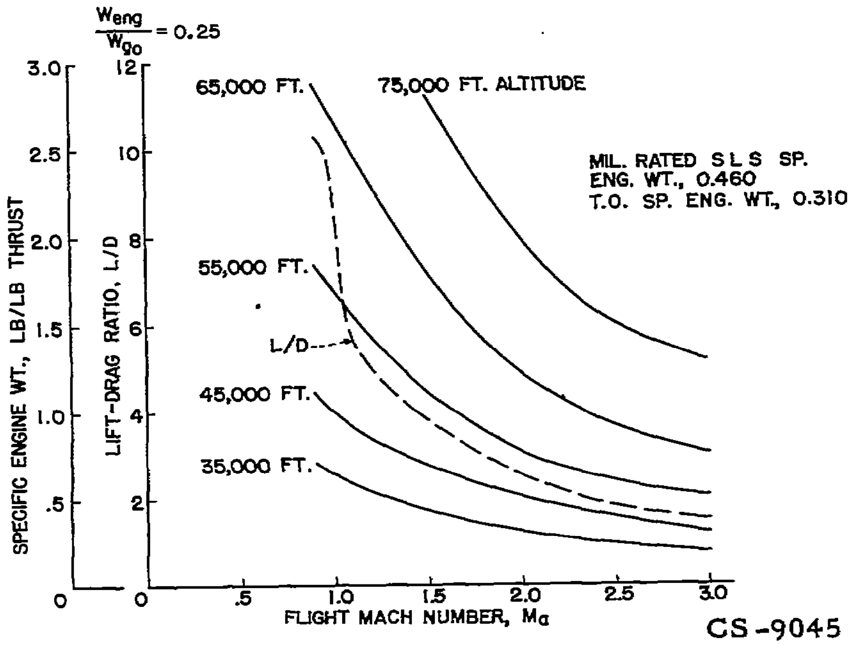

Figure 11. - Effect of airplane speed and altitude on specific engine weight and on airplane L/D.

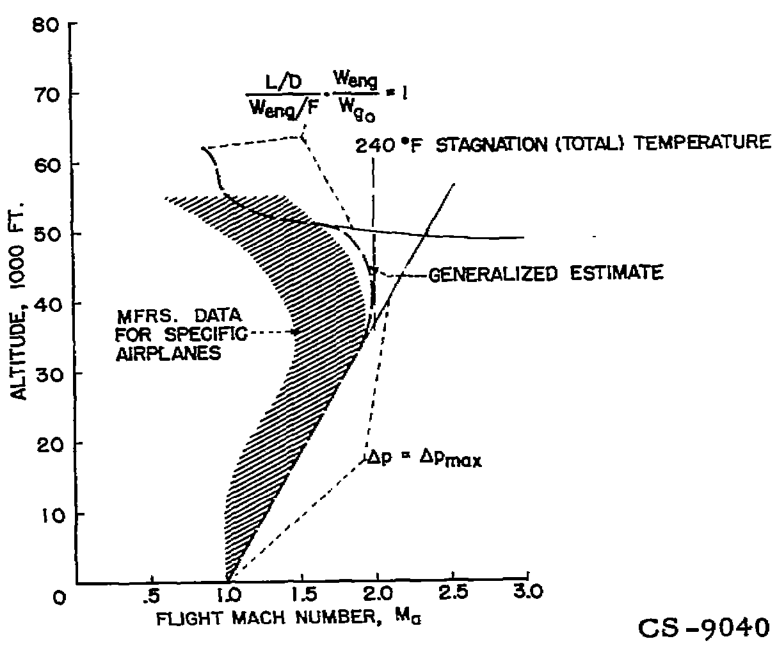

Figure 12. - Effect of airframe and engine [redacted]er altitude and speed limits.

[redacted]

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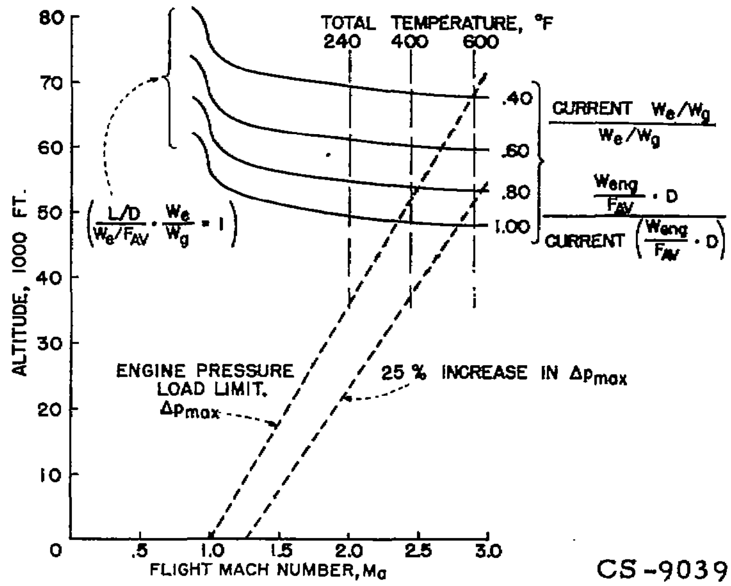

Figure 13. - Effect on fighter altitude and speed limits of decreasing engine specific weight or airplane drag or of increasing allowable engine hoop stress.

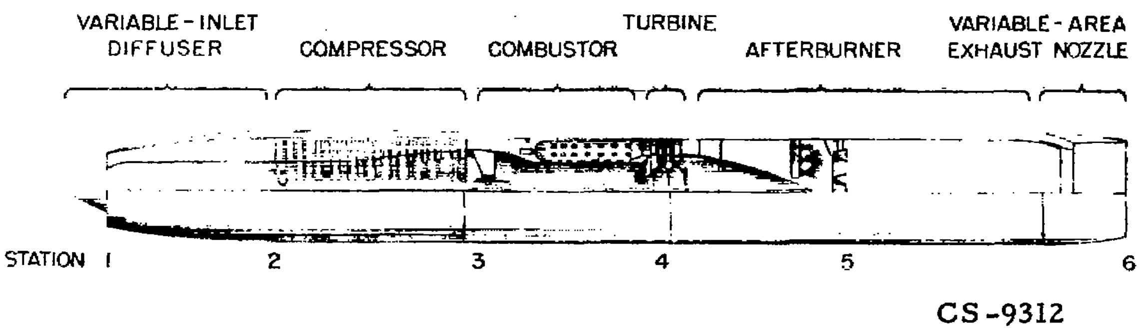

Figure 14. - Diagrammatic sketch of turbojet engine.

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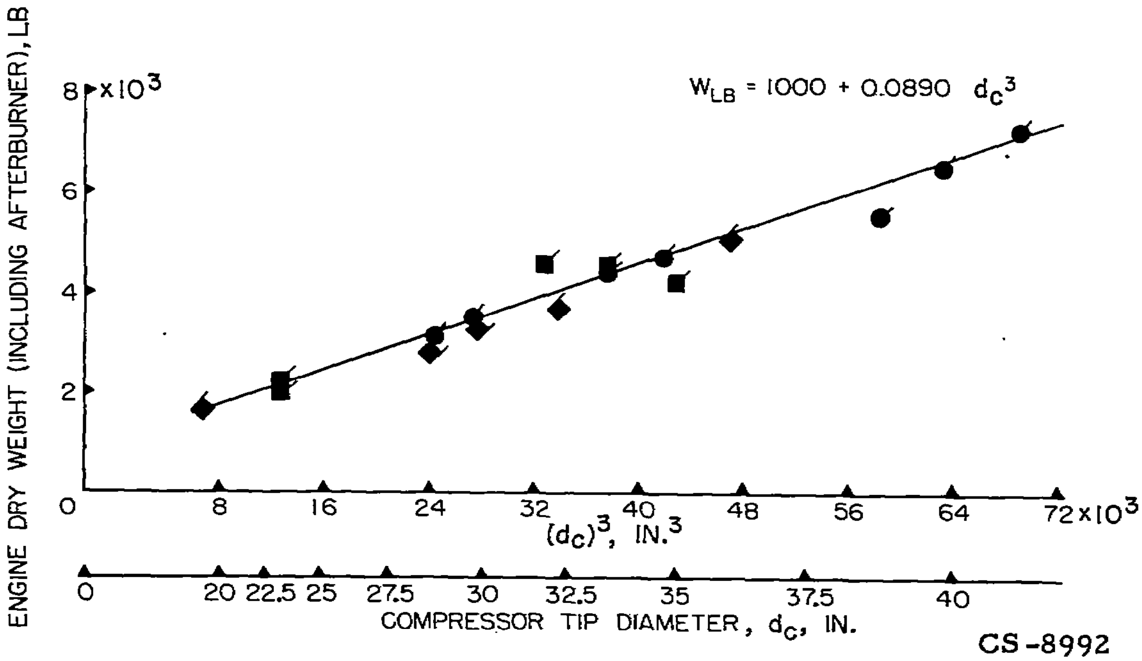

Figure 15. - Relation between compressor tip diameter and engine weight.

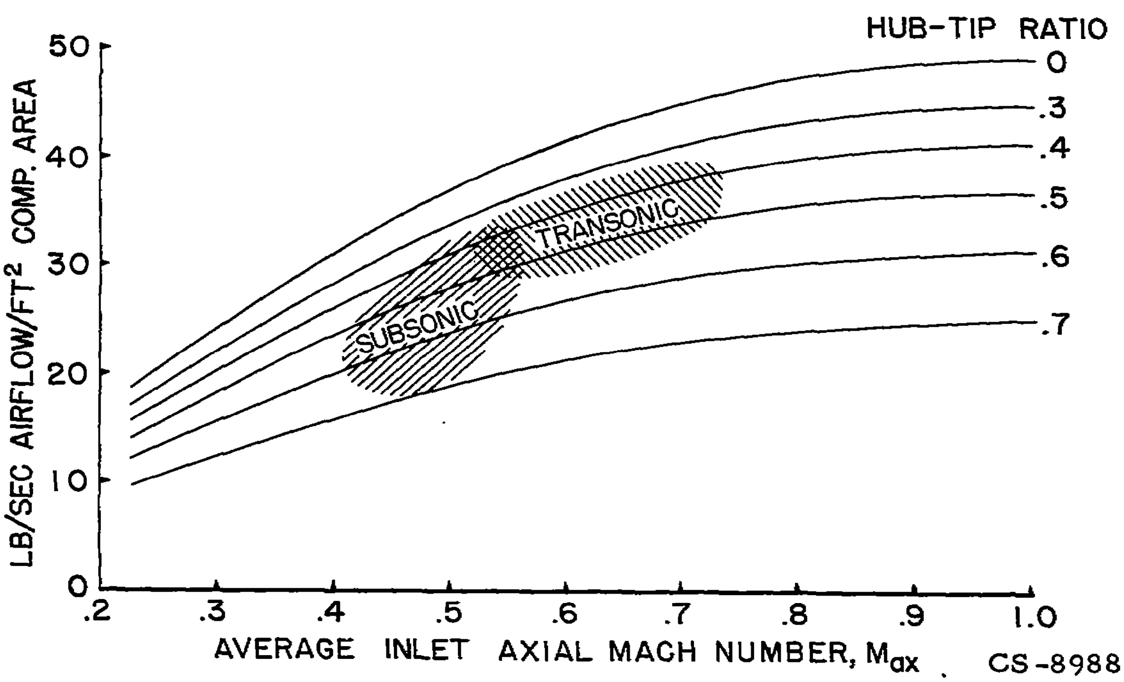

Figure 16. - [redacted] air-handling capacity.

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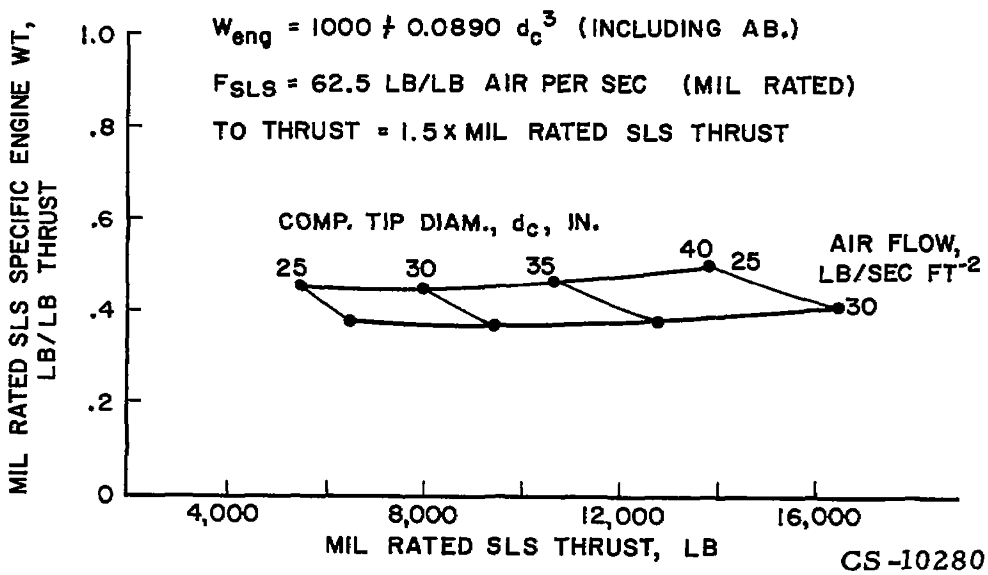

Figure 17. - Effect of engine thrust on specific engine weight.

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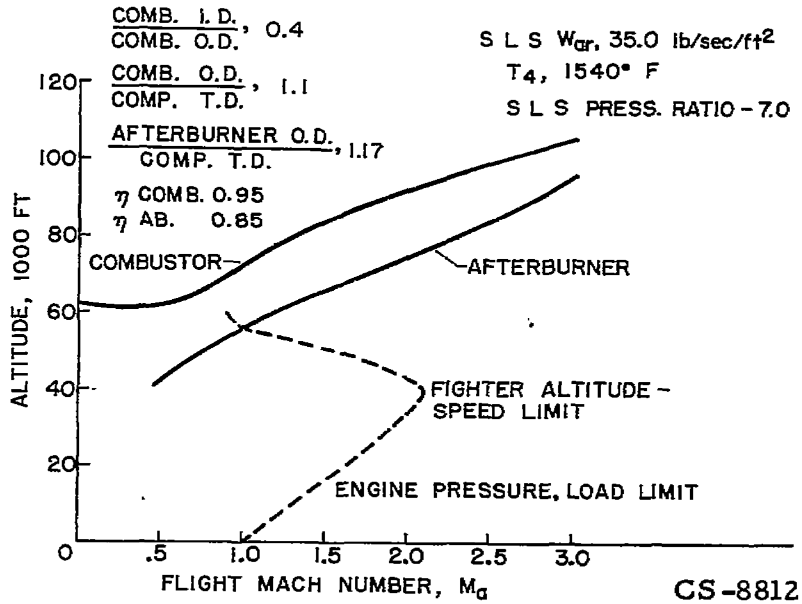

Figure 18. - Limits on airplane performance imposed by combustor performance.

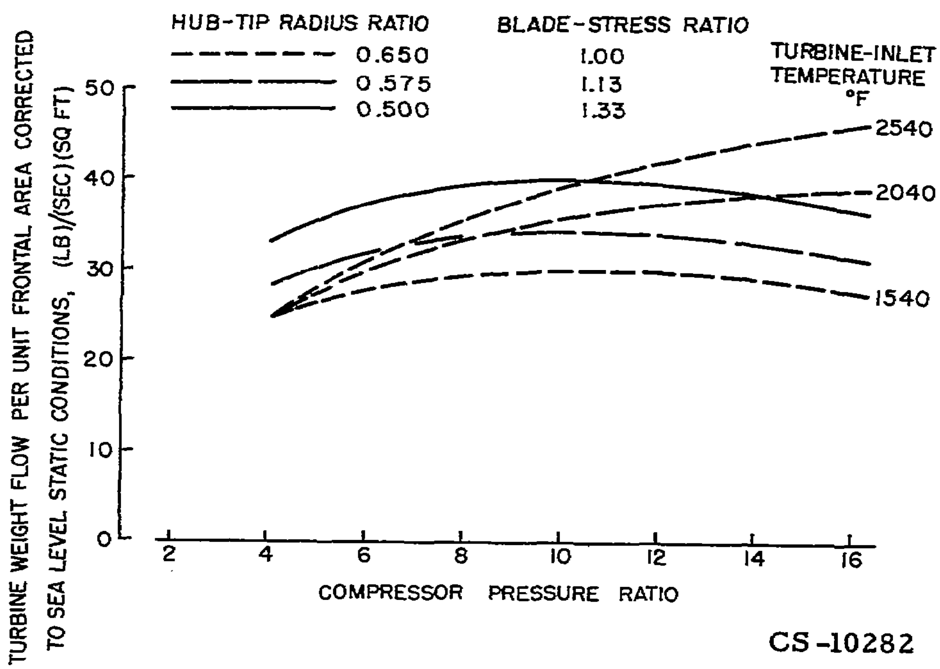

Figure 19. - Effects of engine and turbine design on turbine air handling capacity for flight Mach number of 1.8. Altitude 35,000 feet and above.

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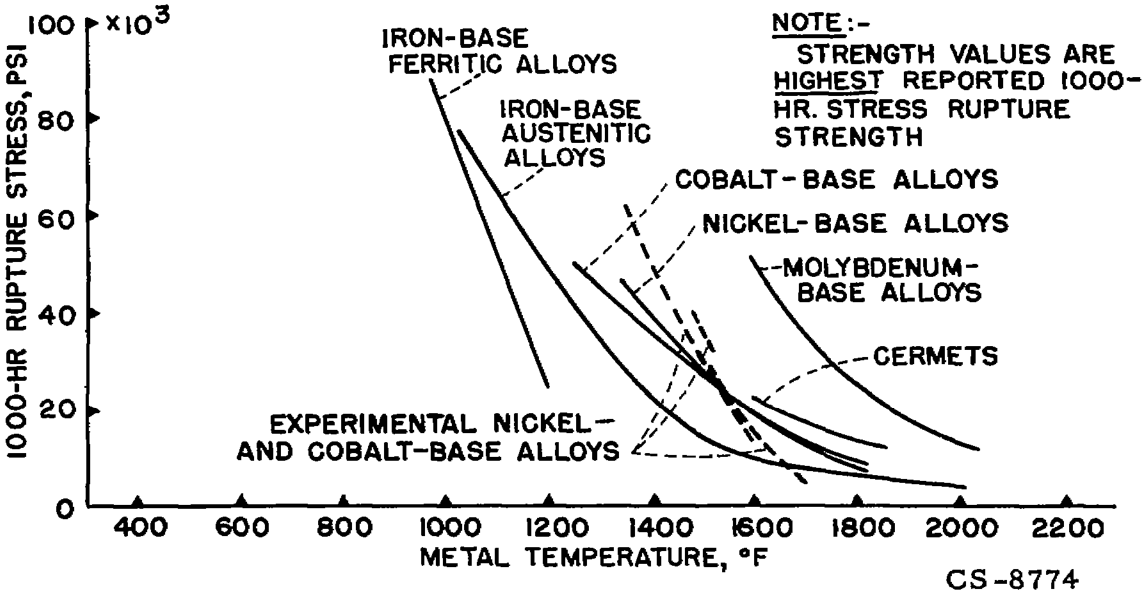

Figure 20. - Effect of temperature on 1000-hour rupture stress for turbine blade alloys. Data (except dashed curves) from General Electric Company.

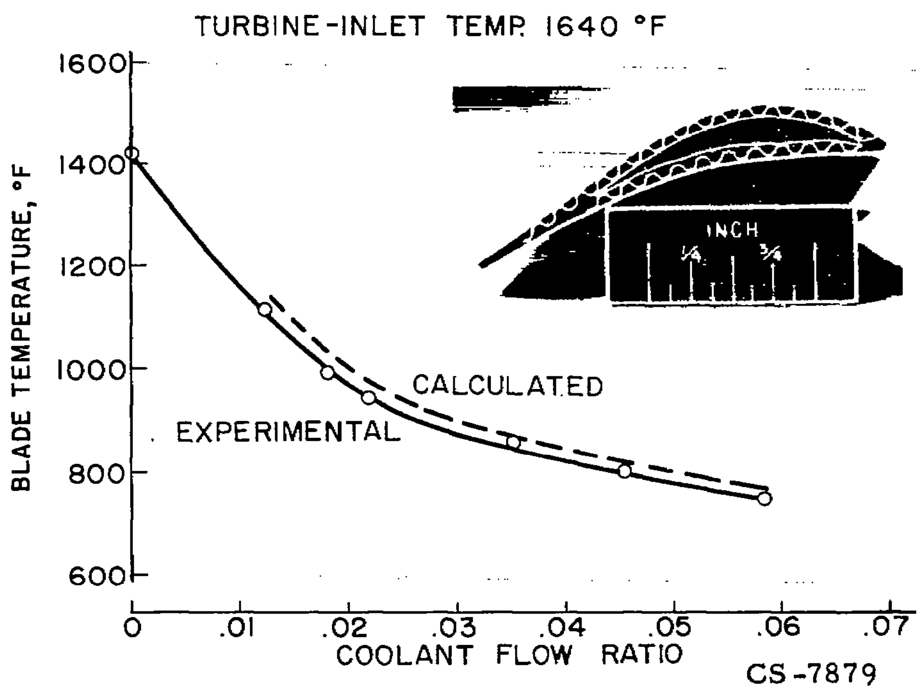

Figure 21. - Comparison of experimental and analytical blade temperatures.

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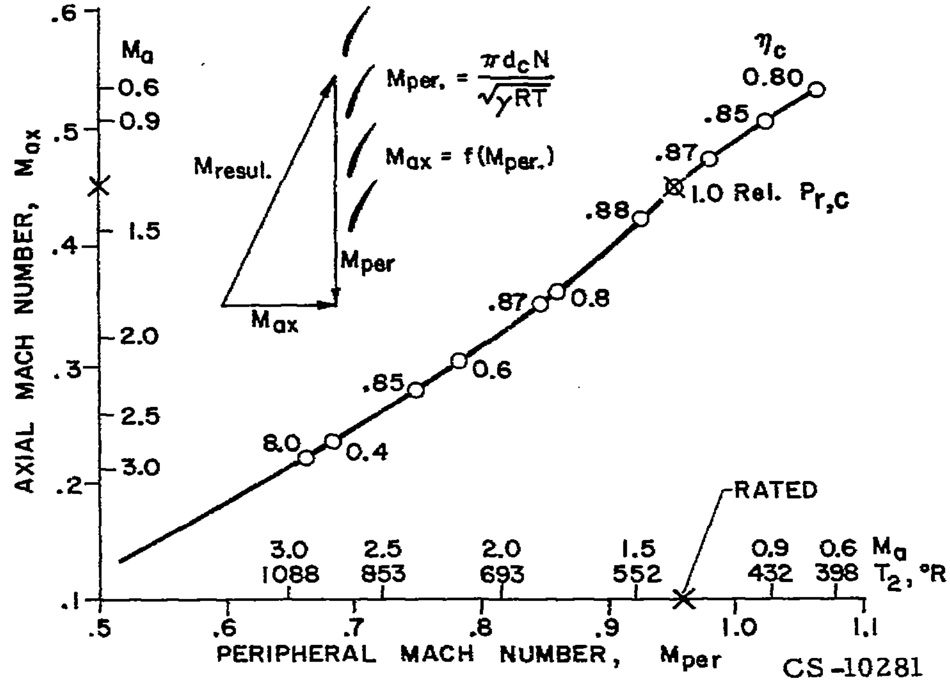

Figure 22. - Effect of airplane speed at altitudes above 35,000 feet, at rated engine speed, on compressor performance.

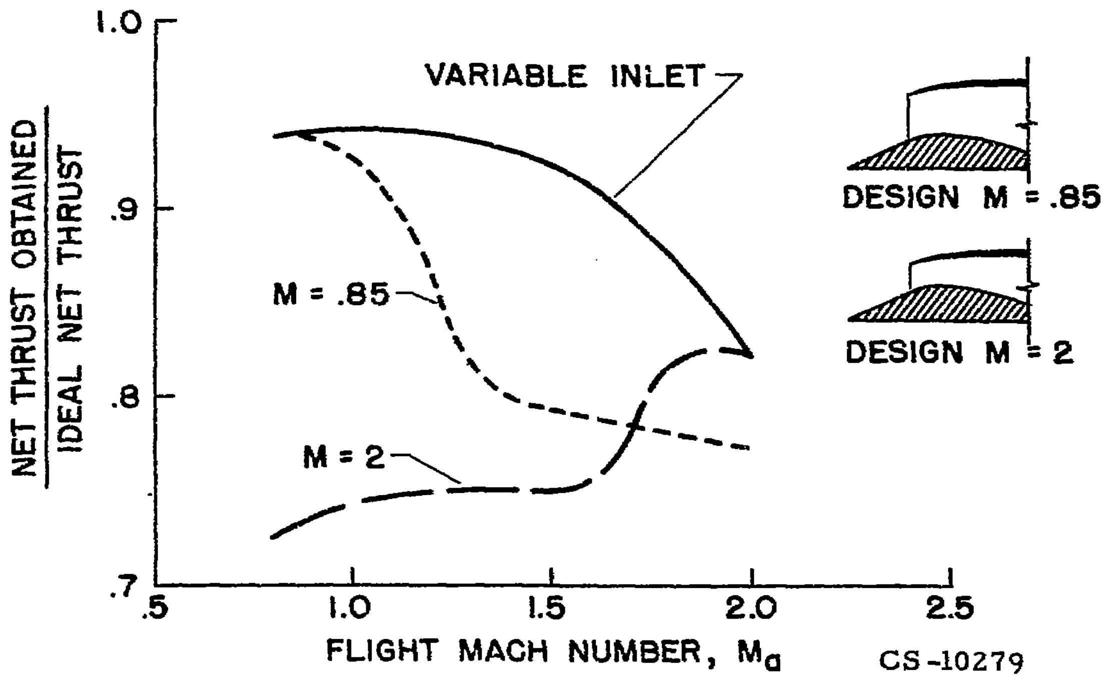

Figure 23. - Effect of fixed and variable inlet diffusers on net thrust.

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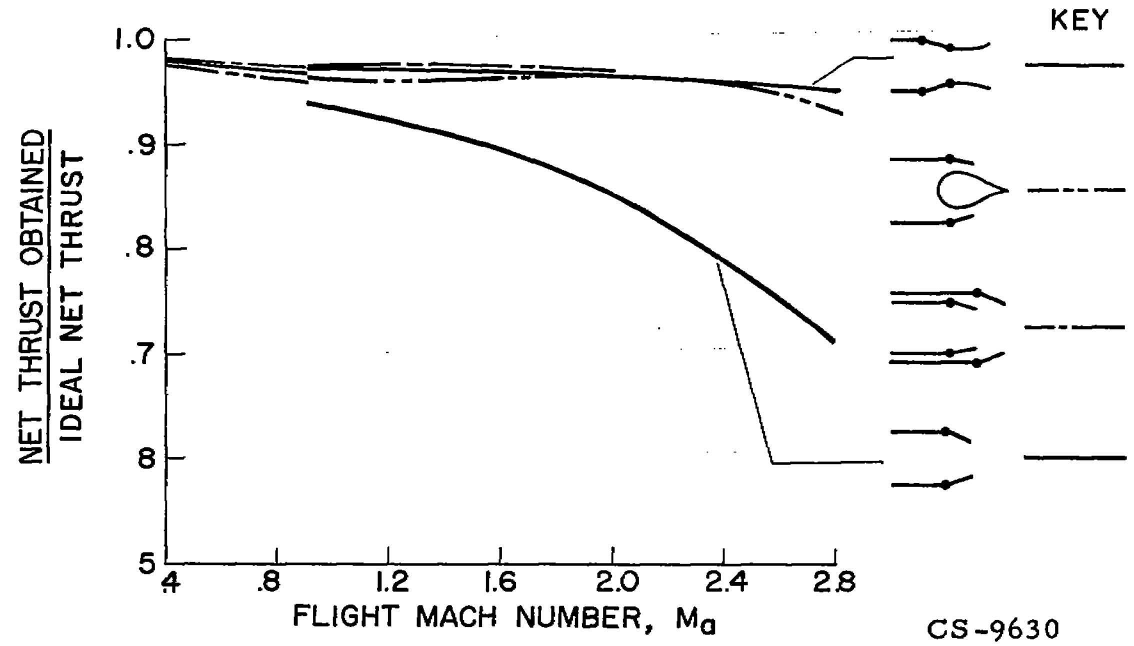

Figure 24. - Comparison of net thrusts using various exhaust nozzles. Data from references 4 to 8.

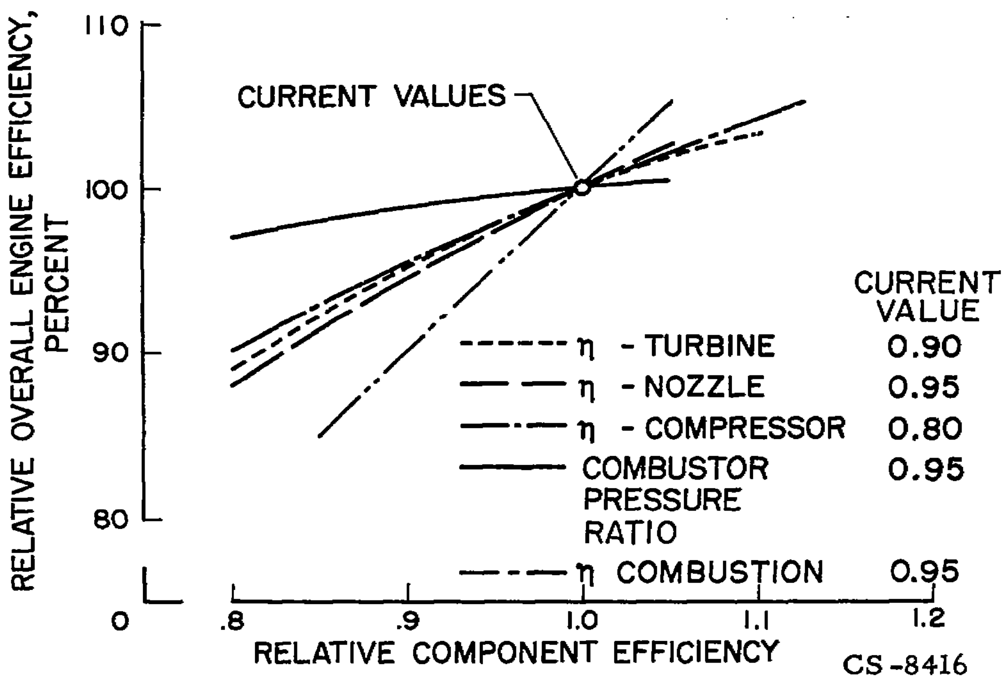

Figure 25. - Effect of component efficiencies on over-all engine efficiency.

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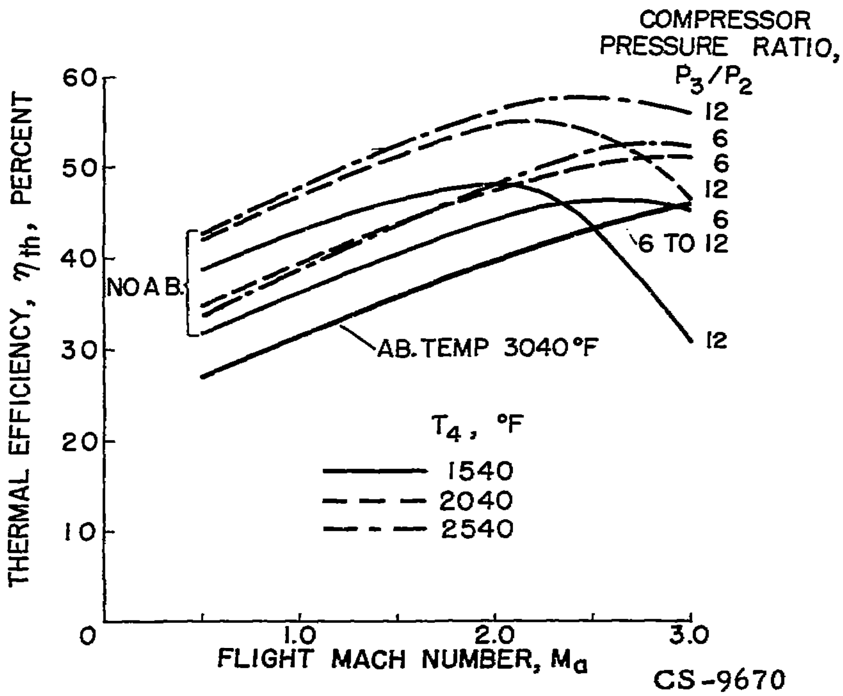

Figure 26. - Relation of airplane speed to engine thermal efficiencies. Altitude 35,000 feet and above.

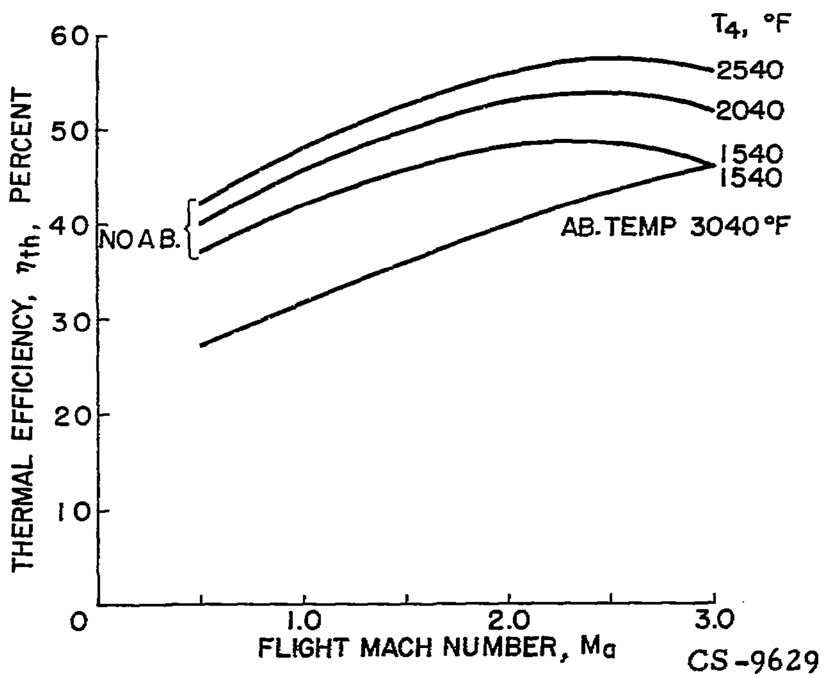

Figure 27. - Relation of airplane speed to engine thermal efficiencies with optimum compressor pressure ratio. Altitude 35,000 feet and above.

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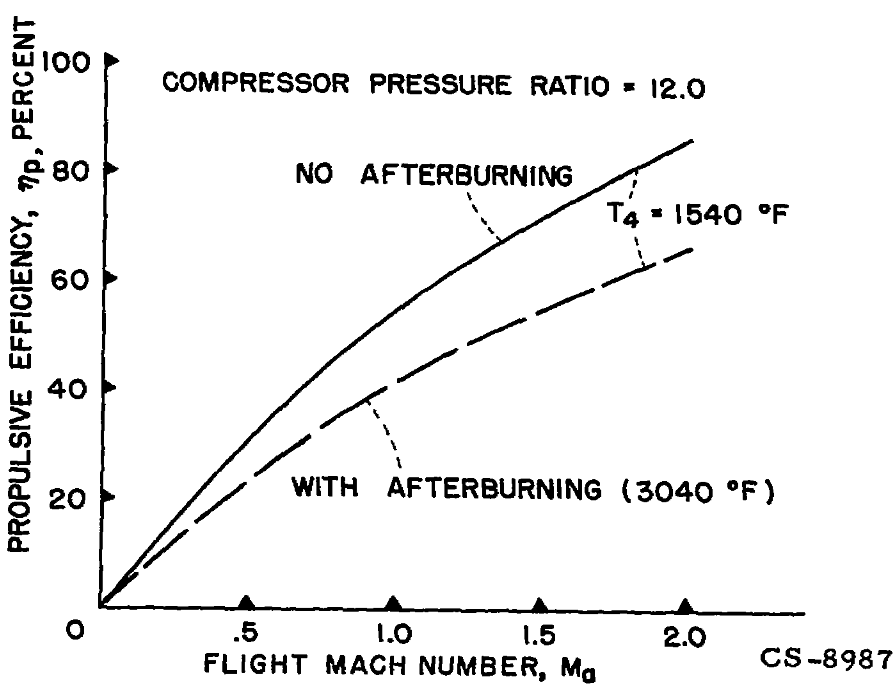

Figure 28. - Effect of flight Mach number on engine propulsive efficiency for altitudes above 35,000 feet.

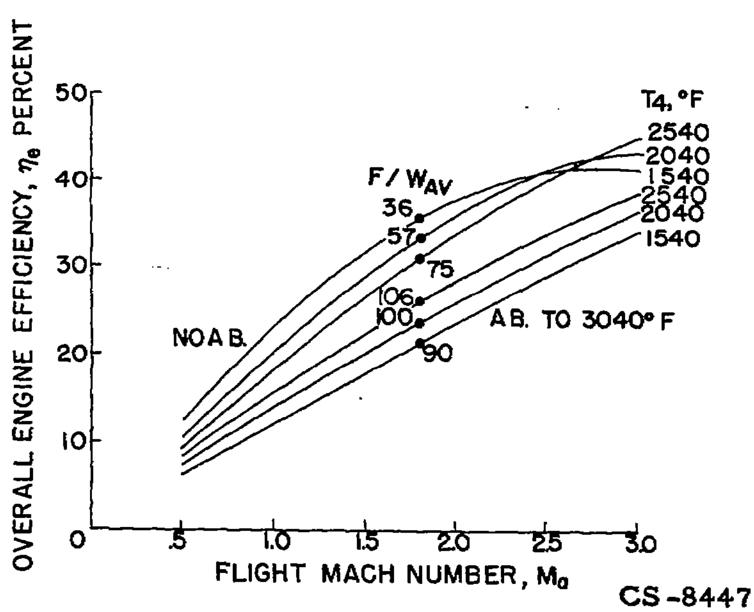

Figure 29. - Effect of flight Mach number on engine efficiencies with and without afterburner. Altitude 35,000 feet and above.

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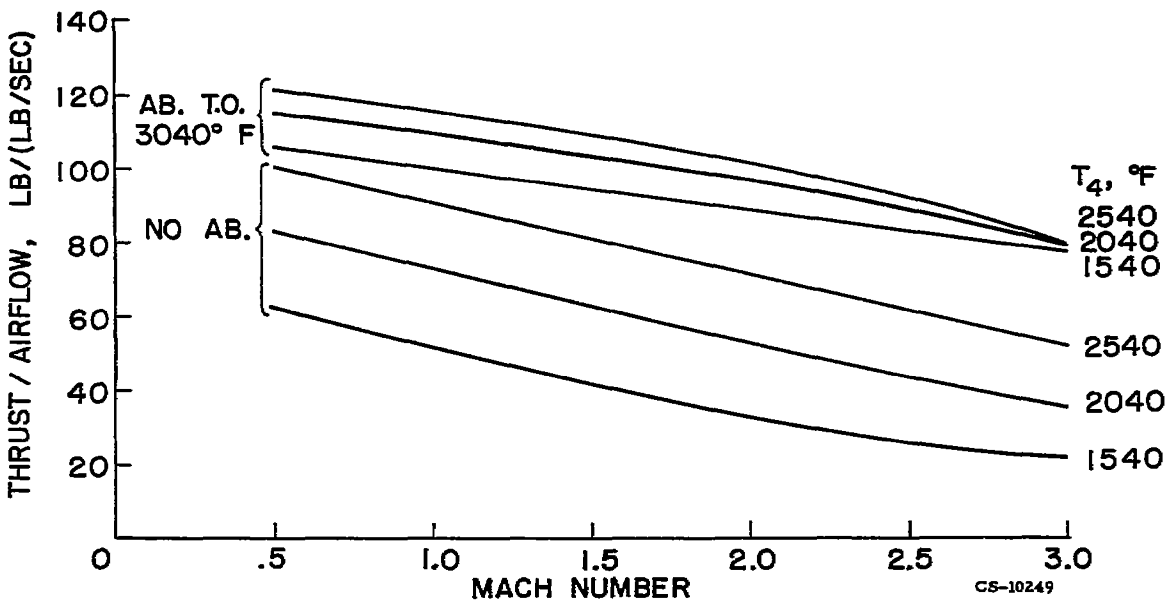

Figure 30. - Effect of flight Mach number on thrust produced per pound of air. Altitude 35,000 feet and above.

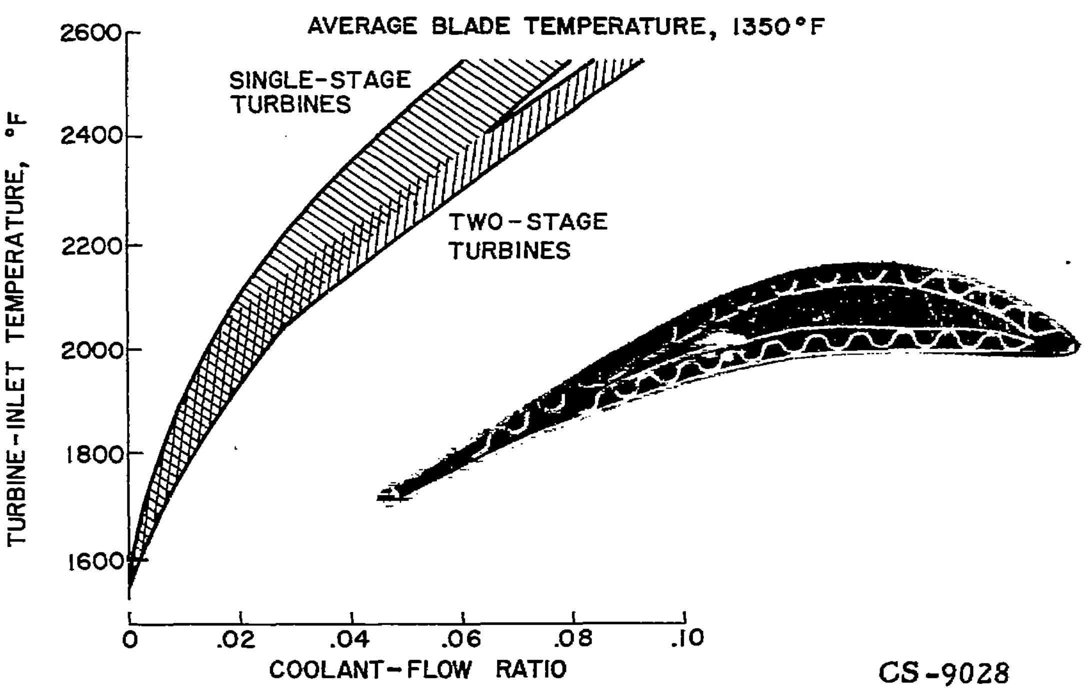

Figure 31. - Coolant-flow requirements for turbines [redacted] air-cooled blades.

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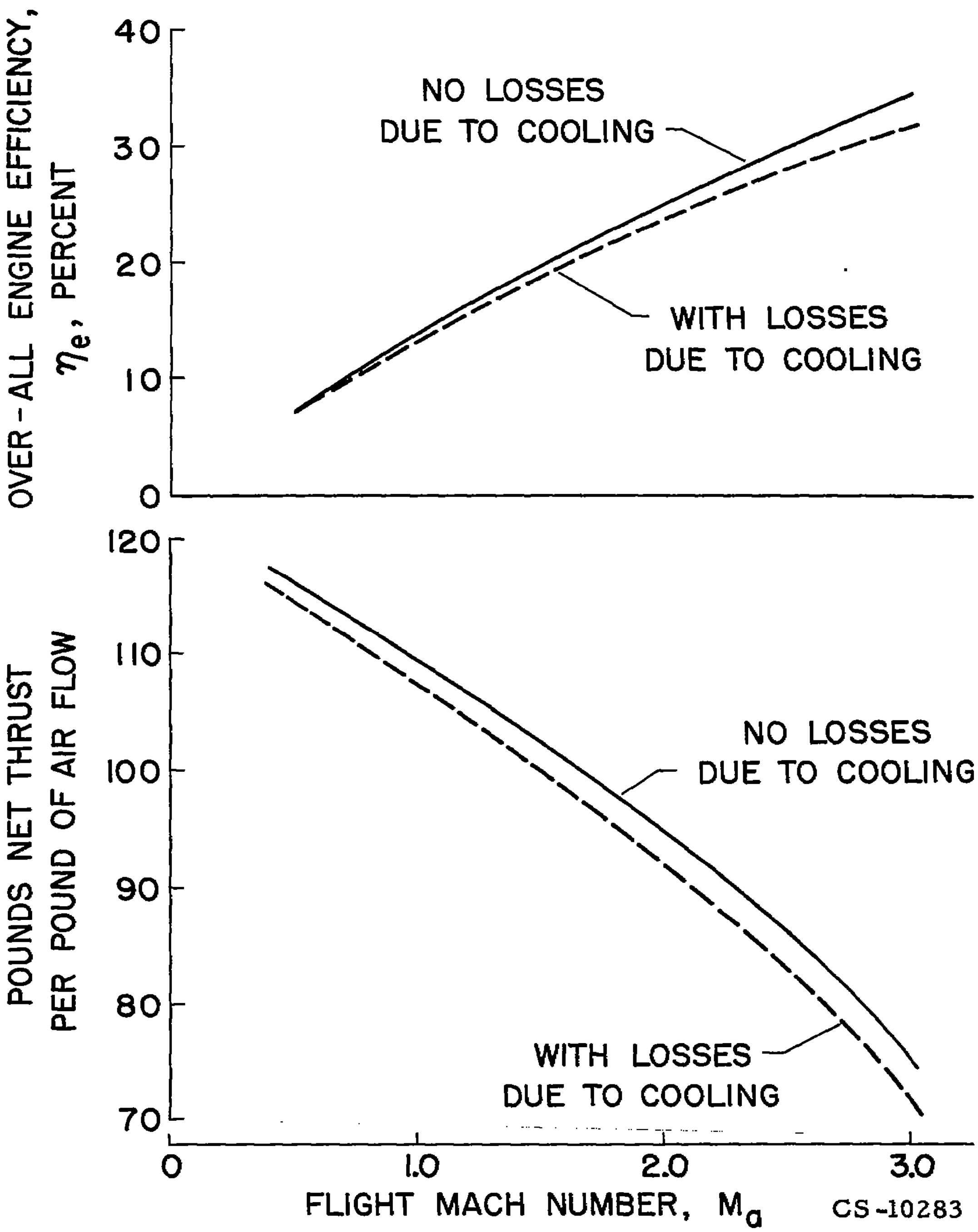

Figure 32. - Estimated turbojet engine cooling losses. Turbine-inlet temperature, 2040° F; afterburner temperature, 3040° F.

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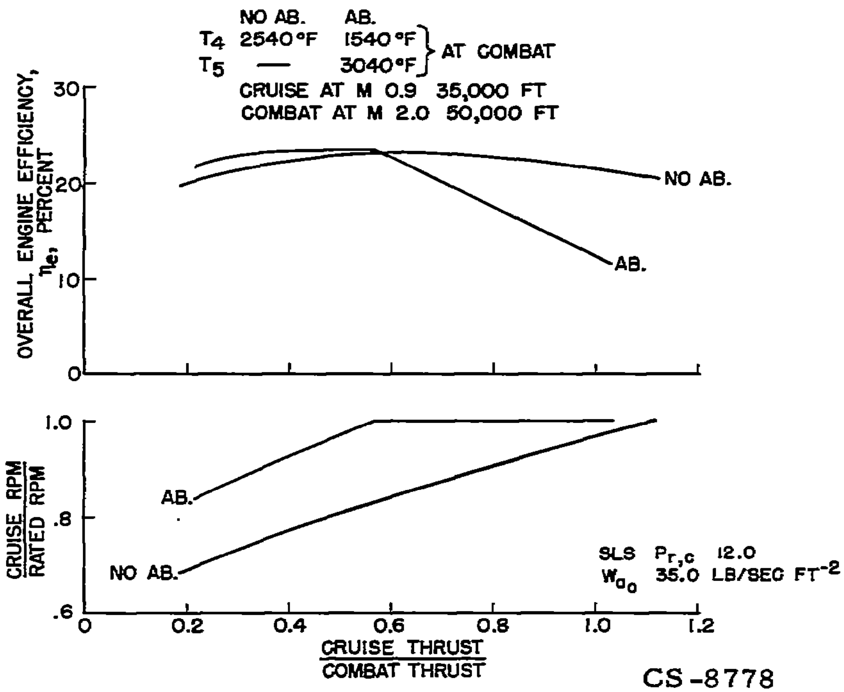

Figure 33. - Effect of afterburner and turbine-inlet temperature on engine efficiency at cruise condition.

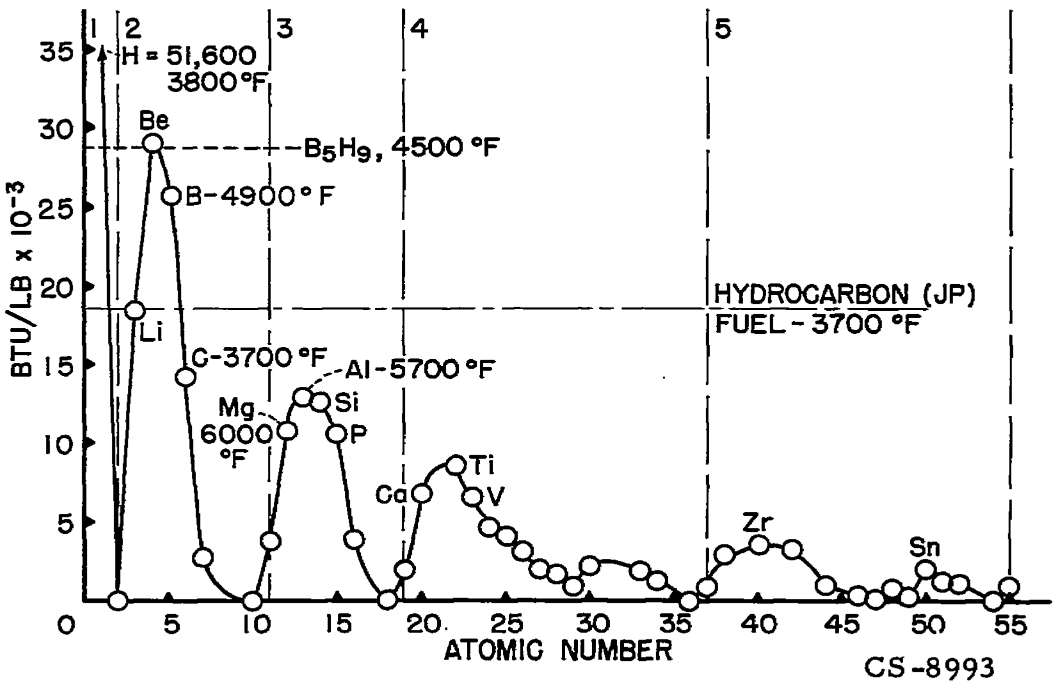

Figure 34. - Heat of [redacted]

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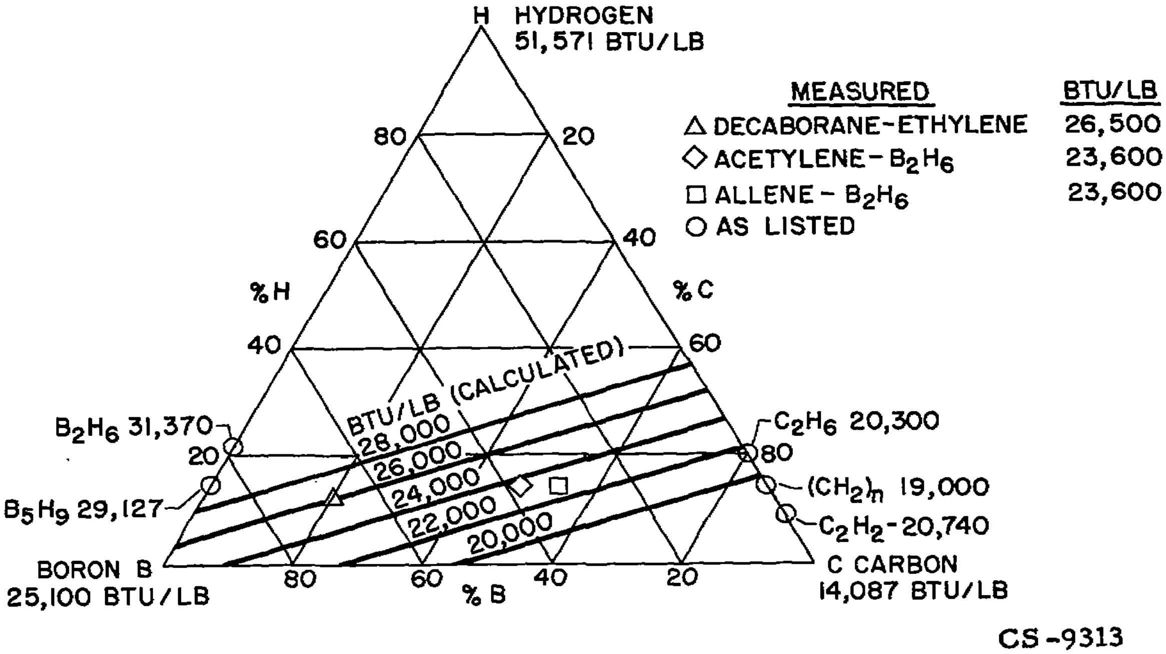

Figure 35. - Heat of combustion versus percent hydrogen, carbon, and boron.

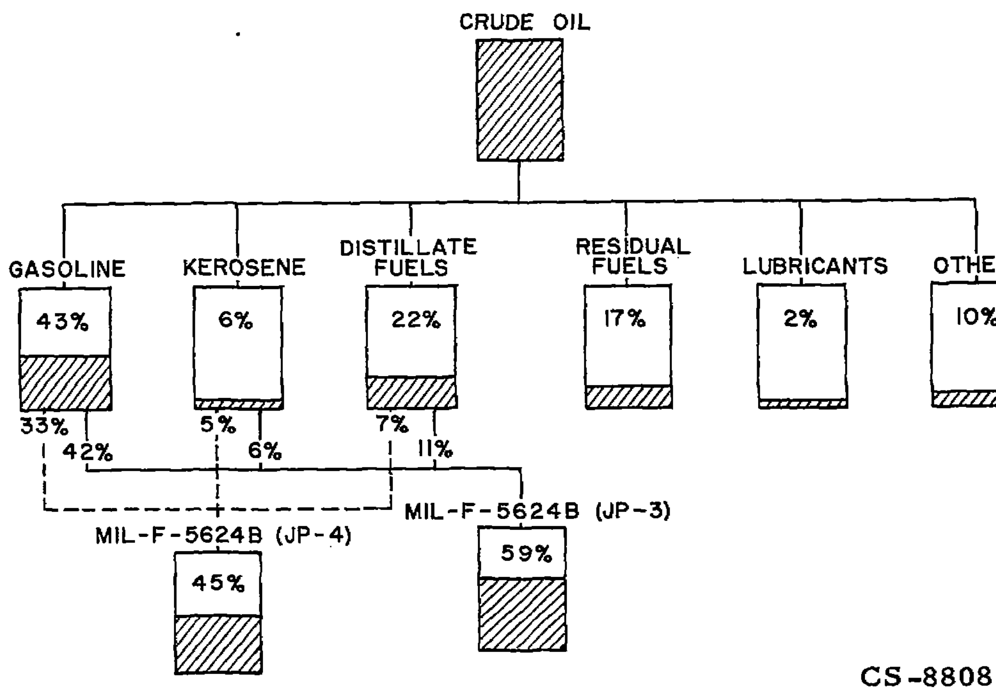

Figure 36. - Availability of fuels from petroleum.

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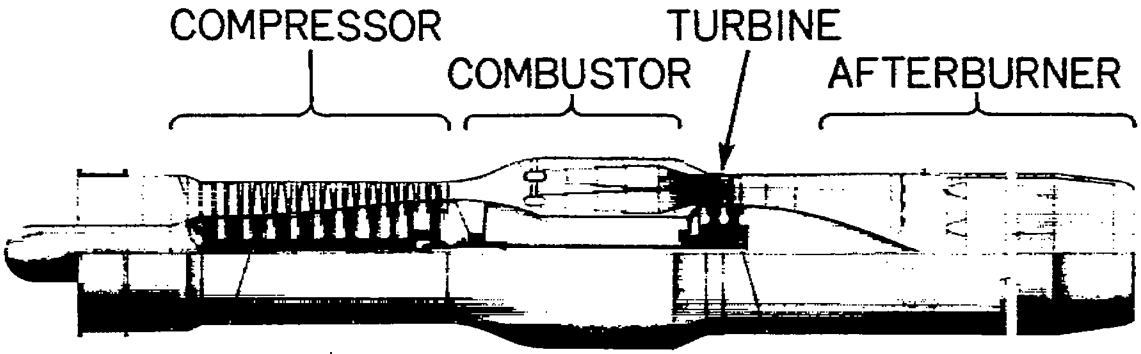

COMPONENT	% ALLOYING ELEMENT					
	Co	Cb	Cr	Ni	Mo	W
COMPRESSOR ASSEMBLY			25			
COMBUSTOR "			25	45		
TURBINE "	75	100	30	20	100	40
AFTERBURNER "	25		20	35		60

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Figure 37. - Distribution of strategic materials in a representative turbojet engine.

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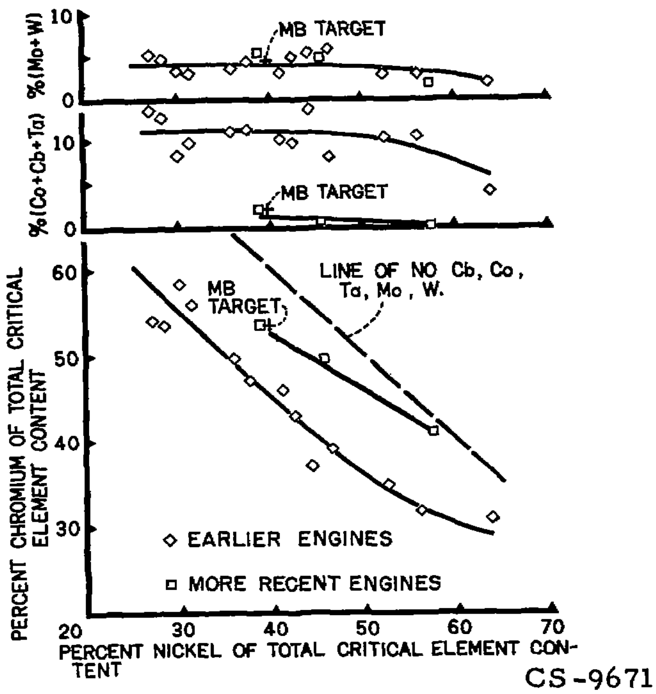

Figure 38. - Ratio of strategic elements on finished weight basis in turbojet engines.

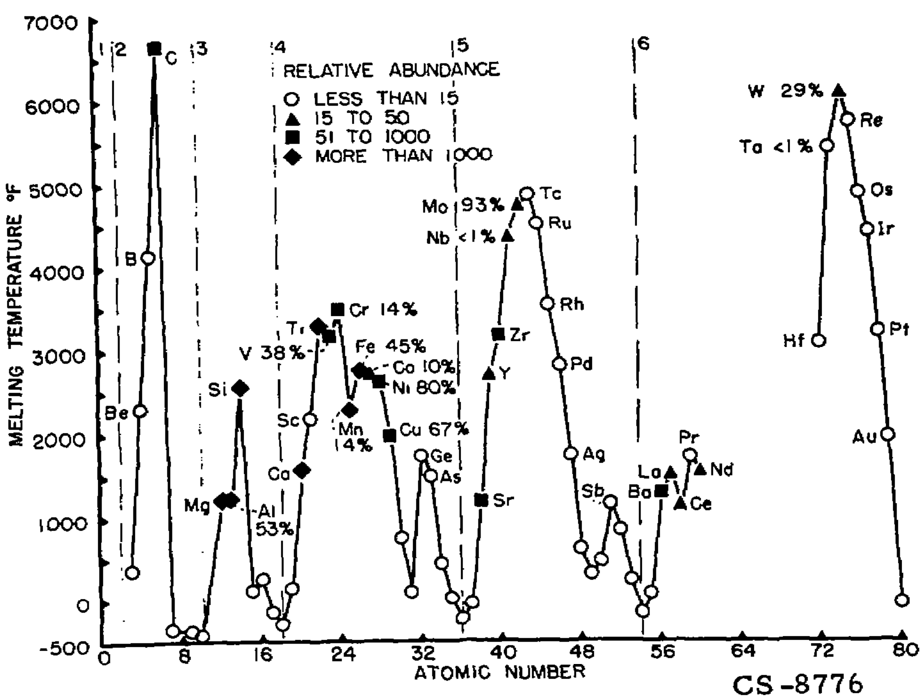

Figure 39. - Melting temperature and relative abundance of the elements.

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Engines, Turbojet 3.1.3

Fuels - Turbine Engines 3.4.3.2

Combustion - Turbine Engines 3.5.2.2

Compressors - Mixed Flow 3.6.1.3

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Rothrock, Addison M.

TURBOJET PROPULSION-SYSTEM RESEARCH AND THE RESULTING EFFECTS ON AIRPLANE PERFORMANCE

Abstract

Airplane performance is analyzed to relate the effects of variations in airplane weight distribution, lift-drag ratio, engine efficiency, specific weight, temperature limitations, and fuel heat of combustion. Possible improvements in turbojet propulsion systems are discussed in connection with research information available in these areas. Fuel availability and the relation of critical materials to over-all engine production is discussed.

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